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Monthly Progress Report

Contract NAS 3-14354

HYDROGEN - OXYGEN HIGH P_c APS ENGINES

Report 14354-M-11

4 October 1971

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Report 14354-M-11

HYDROGEN-OXYGEN HIGH P_C APS ENGINES

NAS 3-14354

Period Ending 30 September 1971

L. Schoenman

4 October 1971

Design & Analysis Department
Aerojet Liquid Rocket Company
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Prepared for
NASA-Lewis Research Center
Cleveland, Ohio 44135

AEROJET LIQUID ROCKET COMPANY
SACRAMENTO, CALIFORNIA • A DIVISION OF AEROJET-GENERAL

FOREWORD

The purpose of this contract is the development of a comprehensive technology base for high performance, long life, gaseous hydrogen-gaseous oxygen rocket engines suitable for the Space Shuttle APS. Significant goals in thruster design are a 50-hour firing life over a 10-year period, with up to 10^6 restarts, and single firings up to 1000 sec.

The program was initially structured as two parallel efforts: one directed toward high pressure (100 to 500 psia) systems and the other toward low pressure (10 to 20 psia) systems. Nominal engine thrust in each case is 1500 lb. Initial program tasks were devoted to the analytical evaluation and screening of injector and cooled thrust chamber concepts for both pressure levels. This was followed by closely paralleled but separate experimental evaluations of low and high pressure injectors and ignition devices. Recommendations of specific injector and igniter designs have been made for both pressure levels as a result of these tests.

As these parallel efforts were about to enter the cooled chamber fabrication phase, the program was redirected to apply additional emphasis on the high P_c technology with a revised schedule on propellant inlet temperatures. Activities on the low pressure phase were terminated by a stop work order, which eliminated the requirements for a portion of the injector testing and all of the low P_c cooled chamber fabrication, durability and pulse testing. The program's resources originally planned for these activities have been reallocated to expand design and test efforts related to the lower temperature gaseous propellants. The high P_c technology effort is now in, the full 40:1 nozzle/thrust chamber, assembly, test phase.

Mr. L. Schoenman, project manager for the high pressure phase, reports to Dr. R. J. LaBotz, who is program manager of all ALRC APS thruster programs. The NASA Lewis Research Center program manager is Mr. J Gregory.

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TABLE LIST

Table I Task X Pulse Test Summary, Series 1680-D02-05

I. PROGRAM OBJECTIVES

The primary objective of this contract is to generate a comprehensive technology base for high performance gaseous hydrogen-gaseous oxygen rocket engines suitable for the Space Shuttle Auxiliary Propulsion System (APS). Durability requirements include injector and thrust chamber designs capable of 50 hours of firing life over a 10-year period with up to 10^6 pulses and single firings up to 1000 sec. These technical objectives are being accomplished and reported upon in a 28-task program summarized below. The first 10 tasks relate to high pressure APS engines, parallel tasks XI through XX relate to low pressure APS engines, and task XXI is a common reporting task. The additional tasks are for the expanded High P_c Low Temperature Program.

Task Titles	High P_c Task		Low P_c Task
	Amb. Prop.	Low Temp Prop.	
Injector analysis and design	I*	XXII*	XI*
Injector fabrication	II*	XXIII*	XII*
Thrust chamber analysis and design	III*	XXIV*	XIII*
Thrust chamber fabrication	IV*	XXV	XIV*
Ignition system analysis and design	V*	--	XV*
Ignition system fabrication and checkout	VI*	--	XVI*
Propellant valves preparation	VII*	--	XVII*
Injector tests	VIII*	XXVI*	XVIII*
Thrust chamber cooling tests	IX*	XXVII	XIX*
Pulsing tests	X	XXVIII	XX*

Common Task

Reporting requirements XXI

*Completed tasks for revised program.

TABLE I

TASK X PULSE TEST SUMMARY
Series 1680-D02-05

Test No.	Date	Function	No. of Pulses	Pulse Width, sec	Off-Time	Results
001		Anemometer calib. (Ox.)	4 units			One unit lost in handling.
002		Anemometer calib. (Fuel)	4 units			Four calibrated units obtained.
003	9/16	Igniter Only Sea Level	Many	.200	Variable	Electrical problems produced non repeatable igniter ignitions, changed spark plug, box & cables; increased power to .02 joules.
003a	9/17	.010 fuel lead	20	.20	.30 sec	Repeatable ignition.
b	9/17	Simultaneous fuel and ox.	20	.20	.30	Satisfactory ignition very repeatable
c		.010 fuel lag	20	.20	.30	Ignition in all cases but not as repeatable as simultaneous or fuel lead.
	Selected .010 fuel lead for igniter start					
d	9/17	Altitude test 100,000 ft	20	.20	.30	Reproducible igniter ignition with fuel lead.
004	9/17	Igniter/engine sequencing .020 sec igniter lead	1	.2	--	Normal thrust chamber ignition - no overpressure
005	9/17	.020 sec igniter lead	1	.2	--	Repeat of above results
006	9/17	Reduce igniter lead	1	.2	--	Error in sequence change - no ignition
007	9/17	Correct error igniter lead,120 sec	1	.2	--	Smooth ignition
008*	9/20	Final sequence adjustment	1	.2	10 min	Satisfactory ignitions in all cases. Postfire
009		variable off-times	1	.2	7 min	inspection showed erosion of tip of film cooling sleeve.
010			1	.2	6 min	Damage attributed to 80 igniter only firings test 003 a-d.
011			2	.2	.3, 120 sec	
012*			5	.2	.3 sec	
013*	9/23	Recheck sequence with repaired chamber	2	.2	.3	Ignition and coolant flow satisfactory
014*		.010 sec fuel valve advance .010 igniter fuel lead LFV to LOV = .02 sec	2	.2	.3	Ignition O.K., sequence too slow; ignition delay in igniter .02 sec.
	Established sequence initiated rapid pulsing					
015*	9/23	Performance & thermal characteristics for 2 cps	100	.2	.3	Steady state temperatures attained; all pulses repeatable
016*	9/23	Repeat of above for additional cycles	110	.2	.3	Computer shut down at 110 pulses due to failure to achieve thrust. Ignition normal, low thrust.
017	9/24	Investigate reason for				
018		low thrust	1	.2	--	See discussion
019	9/27	Change TCV sequence to .005 sec ox. lead	1	.2	20 min	Smoother start - no overpressure in oxidizer manifold
020	9/27	Repeat of above performance,	2	.2	.3	Pulses repeatable
021	9/27	thermal & durability	266	.2	.3	Pulses repeatable, low thrust shutdown on pulse 266 (no igniter ignition).
022	9/27	Single pulse of above sequence	1	.2	--	Normal ignition and pulse.

TABLE I (cont.)

Test No.	Date	Function	No. of Pulses	Pulse Width, sec	Off-Time	Results
023	9/27	Continuation of durability testing	259	.2	.3	2 cps 40% duty cycle satisfactorily completed.
024	9/27	3.3 cps 40% duty cycle investigation	4	.1	.15	Satisfactory ignition and repeatability.
025*	9/27	Repeat of above with variable off-times	1	.1	--	Normal ignition and repeatability in all tests
026*			1	.1	--	
027*			4	.1	.15	
028*	9/27	Performance repeatability and durability tests	760	.1	.15	Normal ignition and repeatability in all tests
029	9/28	10 cps 35% duty cycle low impulse bits sequence evaluation variable off-time	1	.035	10 min	Normal ignition and repeatability in all tests Minimum off-time between pulses for valve sequencing .065 sec.
030			1	.035	5 min	
031			1	.035	5 min	
032			10	.035	.065 sec	
033			1	.035	3 min	
034			1	.035	7 min	
035			1	.		
036*		Repeatability & durability	100	.035	.065	Normal ignition and repeatability
037*		Continuation of durability testing	280	.035	.065	Fuel valve operation irregular due to pilot valve actuator overheating after approx. 250 pulses
038*		Continuation of durability testing	840	.100	.150	Satisfactory completion of pulse test series with over 2500 pulses on SN-7 injector and 2700 pulses on SN-2 film cooled chamber

* Tests with anemometers

II. PROGRESS BY TASK

A. AMBIENT TEMPERATURE PROPELLANT TASKS

1. Tasks I through IX were completed and reported upon in previous progress reports.

2. Task X - Pulse Testing

Task X pulsing tests were successfully completed during this report period. These activities include the restarting of a single thrust chamber assembly in excess of 2500 times without intermediate service or disassembly. This assembly contained: (1) a 25-lbf spark igniter, (2) a premix "I" triplet injector (SN-7), (3) SN-2 film cooled chamber, and (4) fast response low pressure drop valves. These tests are summarized in Table I and the preliminary results are presented in Appendix A.

B. COLD PROPELLANT TASKS

1. Tasks XXII and XXIII were completed and reported upon earlier in the program.

2. Task XXIV - Cooled Chamber Design

Design activities on cooled thrust chambers were completed during this report period including the complete thermal and structural life analysis of three types of manifolding arrangements for the film cooled chamber. Manifold designs considered are:

a. Non Contacting Manifold and Liner

b. Monowall - Contacting Manifolding and Liner

c. Dual Wall - Contacting Liner and Manifolding for Thermal Isolation

II, B, Cold Propellant Tasks (cont.)

The design which provides the greatest chamber life is the dual wall contacting design with thermal isolation from the propellant. This design provides life capabilities in the manifold and chamber which are comparable with the remainder of the chamber. A discussion of these designs is presented in Appendix B.

3. Task XXV - Cooled Chamber Fabrication

a. Regeneratively Cooled Chamber

As of the close of this report period, SN-3 film cooled chamber had reached the fabrication point where the only remaining operation is to weld on the spun film cooled skirt and complete the thermal instrumentation. In the present configuration, this chamber contains the steel jacket which closes out the coolant channels, all inlet lines, fittings and pressure taps, and all brazed-in gas-side thermocouples.

b. Film Cooled Chamber

As of the close of the report period, SN-3 film cooled chamber, which is a rework of SN-1 unit to incorporate film coolant injection orifices optimum for cold propellants, was complete including all instrumentation. The chamber is currently located in the test area buildup shop in a ready-to-fire condition.

4. Task XXVI - Injector Checkout Tests

Task completed. Results reported in Q-4, M-9, and M-10.

II, B, Cold Propellant Tasks (cont.)

5. Task XXVII - Cooled Chamber Testing

Testing of film and regeneratively cooled chambers optimized for low temperature propellants will be accomplished in the task. Testing of the film cooled chamber is scheduled to start the first week in October.

6. Task XXVIII - Low Temperature Pulse Testing

See discussion in Section IV.

III. WORK DURING NEXT REPORTING PERIOD

Task I through IX - no activity.

Task X - complete pulse test data analysis.

Task XXII through XXIV - no activity.

Task XXV - completion of regen chamber fab and instrumentation.

Task XXVI - no activity.

Task XXVII and XXVIII - complete an integrated program of long duration and pulse testing with cold propellants.

IV. PROBLEM AREAS

No major technical problem areas have been encountered in this program to date. An accumulation of minor problems, for the most part in Task X, the pulse testing, has resulted in budget and schedule problems in the program. The Task X delays include:

(1) Calibration of a new set of higher strength anemometer in the test run position. These are improved versions of those calibrated in May.

IV, Problem Areas (cont.)

(2) Thrust chamber valve inspection and servicing prior to initiation of pulse testing, replacement of stem and seat seals and reallignment of valve actuators.

(3) Igniter electrical problems after buildup with new spark plug.

(4) Damage and rework of the tip of the film cooling ring resulting from igniter only mode pulse testing.

(5) Changing of injectors due to oxidizer diffuser damage resulting from pressure spikes in the oxidizer manifold.

The fiscal difficulties encountered in Task X can be accommodated by integrating the Task XXVIII cold pulse testing with the Task XXVII cooled chamber testing with cold propellants and deleting the requirement for an additional 2500 pulses. A program schedule extension of several weeks will be adequate to allow completion of test activities, in a more efficient manner.

Projected changes in the corporate overhead rates are expected to result in a 2% increase in program costs. These are not reflected in the attached Form 533b.

FORECAST AND CONSUMPTION OF GOVERNMENT-FURNISHED PROPELLANTS

Contract NAS 3-14354

Material	September Monthly Usage	Cumulative	Next Month's Requirements	Next 3-Month Requirements
LO_2 (ton)	20.9	80.2	40	120
LH_2 (lb)	0	7210	0	0
LN_2 (ton)	153	755	150	250
GHe 10^3 (SCF), Bulk	0	99,100	0	0
GHe 10^3 (SCF), Cylinders	0	0	19	7

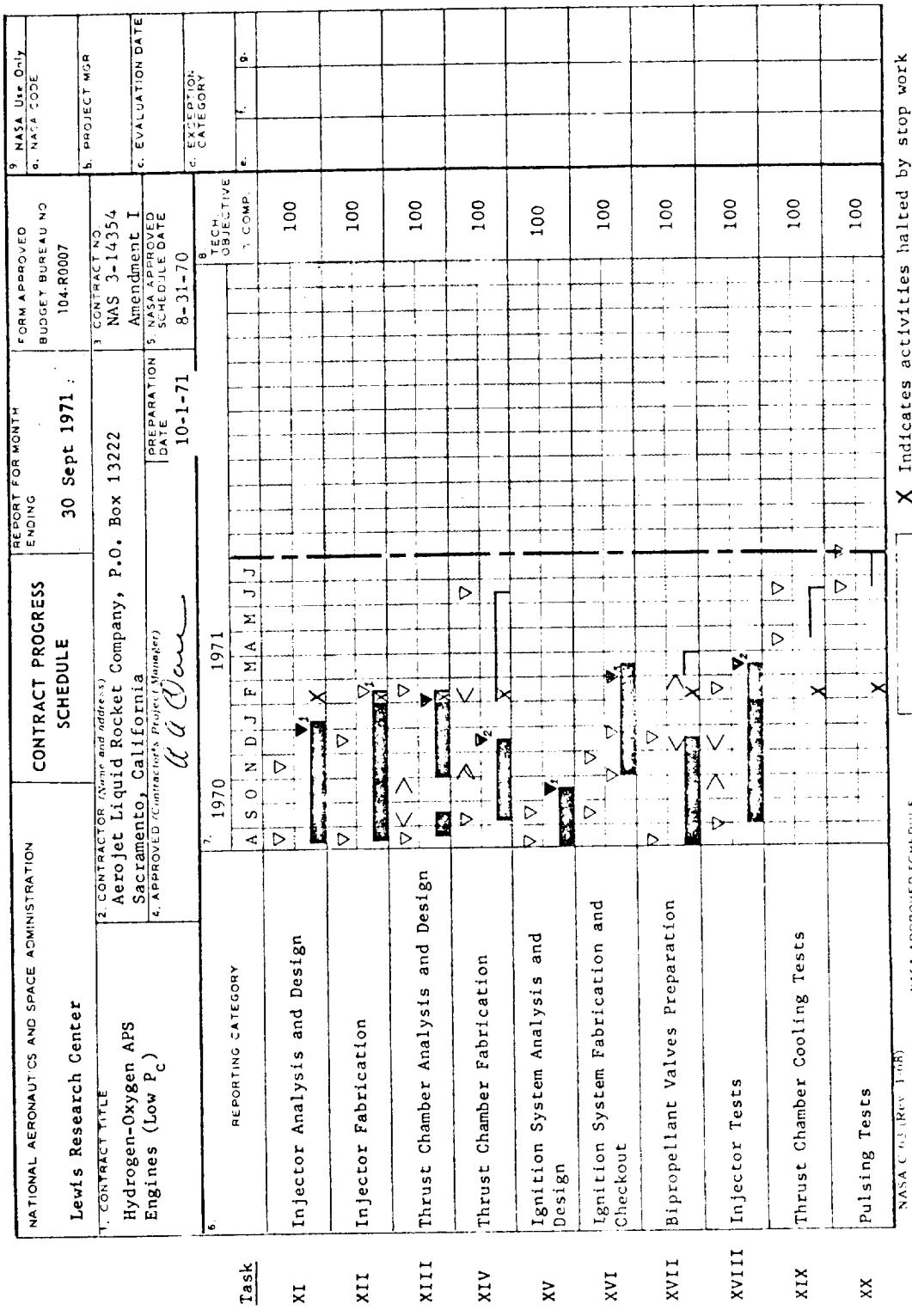
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION	CONTRACT PROGRESS SCHEDULE	REPORT FOR MONTH ENDING	FORM APPROVED BUDGET BUREAU NO	9. NASA Use Only
Lewis Research Center		30 Sept 1971	104-R0007	a. NASA CODE
1. CONTRACT TITLE: Hydrogen-Oxygen APS Engines (High P_c)	2. CONTRACTOR (Name and address): Aerojet Liquid Rocket Co., P.O. Box 13222 Sacramento, California 95813		3. CONTRACT NO: NAS 3-14354	b. PROJECT MGR
	4. APPROVED (Contractor's Project Manager): L Sherman	PREPARATION DATE: 10-1-71	5. NASA APPROVED SCHEDULE DATE: 8-31-70	c. EVALUATION DATE
				d. EXCEPTION CATEGORY

Task	6. REPORTING CATEGORY	7. 1970 J A S O N D / 1971 J F M A M J J A S	8. TECH OBJECTIVE % COMP.	e.	f.	g.
I	Injector Analysis and Design		100			
II	Injector Fabrication		100			
III	Thrust Chamber Analysis and Design		100			
IV	Thrust Chamber Fabrication		100			
V	Ignition System Analysis and Design		100			
VI	Ignition System Fabrication and Checkout		100			
VII	Bipropellant Valve Preparation		100			
VIII	Injector Tests		100			
IX	Thrust Chamber Cooling Tests		100			
X	Pulsing Tests		99			

NASA C-63 (Rev. 1-68)

NASA APPROVED SCHEDULE

CONTRACTOR'S WORKING SCHEDULE



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center

CONTRACT PROGRESS SCHEDULE

REPORT FOR MONTH ENDING: 30 Sept 1971

FORM APPROVED BUDGET BUREAU NO 104-R0007

9. NASA Use Only
a. NASA CODE
b. PROJECT MGR

1. CONTRACT TITLE: Hydrogen-Oxygen APS Engines (Low P_c)

2. CONTRACTOR (Name and address): Aerojet Liquid Rocket Company, P.O. Box 13222 Sacramento, California

3. CONTRACT NO: NAS 3-14354 Amendment I

4. APPROVED (Contractor's Project Manager): W. A. Clem

PREPARATION DATE: 10-1-71

5. NASA APPROVED SCHEDULE DATE: 8-31-70

c. EVALUATION DATE

c. EXCEPTION CATEGORY

Task	6. REPORTING CATEGORY	7. 1970 A S O N D	1971 J F M A M J J	8. TECH. OBJECTIVE % COMP.	e.	f.	g.
XI	Injector Analysis and Design			100			
XII	Injector Fabrication			100			
XIII	Thrust Chamber Analysis and Design			100			
XIV	Thrust Chamber Fabrication			100			
XV	Ignition System Analysis and Design			100			
XVI	Ignition System Fabrication and Checkout			100			
XVII	Bipropellant Valves Preparation			100			
XVIII	Injector Tests			100			
XIX	Thrust Chamber Cooling Tests			100			
XX	Pulsing Tests			100			

NASA C-53 (Rev 1-68) NASA APPROVED SCHEDULE CONTRACTOR'S WORKING SCHEDULE

X Indicates activities halted by stop work order dated 11 February 1971.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION	CONTRACT PROGRESS SCHEDULE	REPORT FOR MONTH ENDING	FORM APPROVED BUDGET BUREAU NO.	9 NASA Use Only
Lewis Research Center		30 Sept 1971	104-R0007	a. NASA CODE
1. CONTRACT TITLE: Hydrogen-Oxygen APS Engines	2. CONTRACTOR (Name and address): Aerojet Liquid Rocket Company, P.O. Box 13222 Sacramento, California 95813		3. CONTRACT NO.: NAS 3-14354	b. PROJECT MGR.
	4. APPROVED (Contractor's Project Manager): R J [illegible]	PREPARATION DATE: 10-1-71	5. NASA APPROVED SCHEDULE DATE: 8-31-70	c. EVALUATION DATE
				d. EXCEPTION CATEGORY

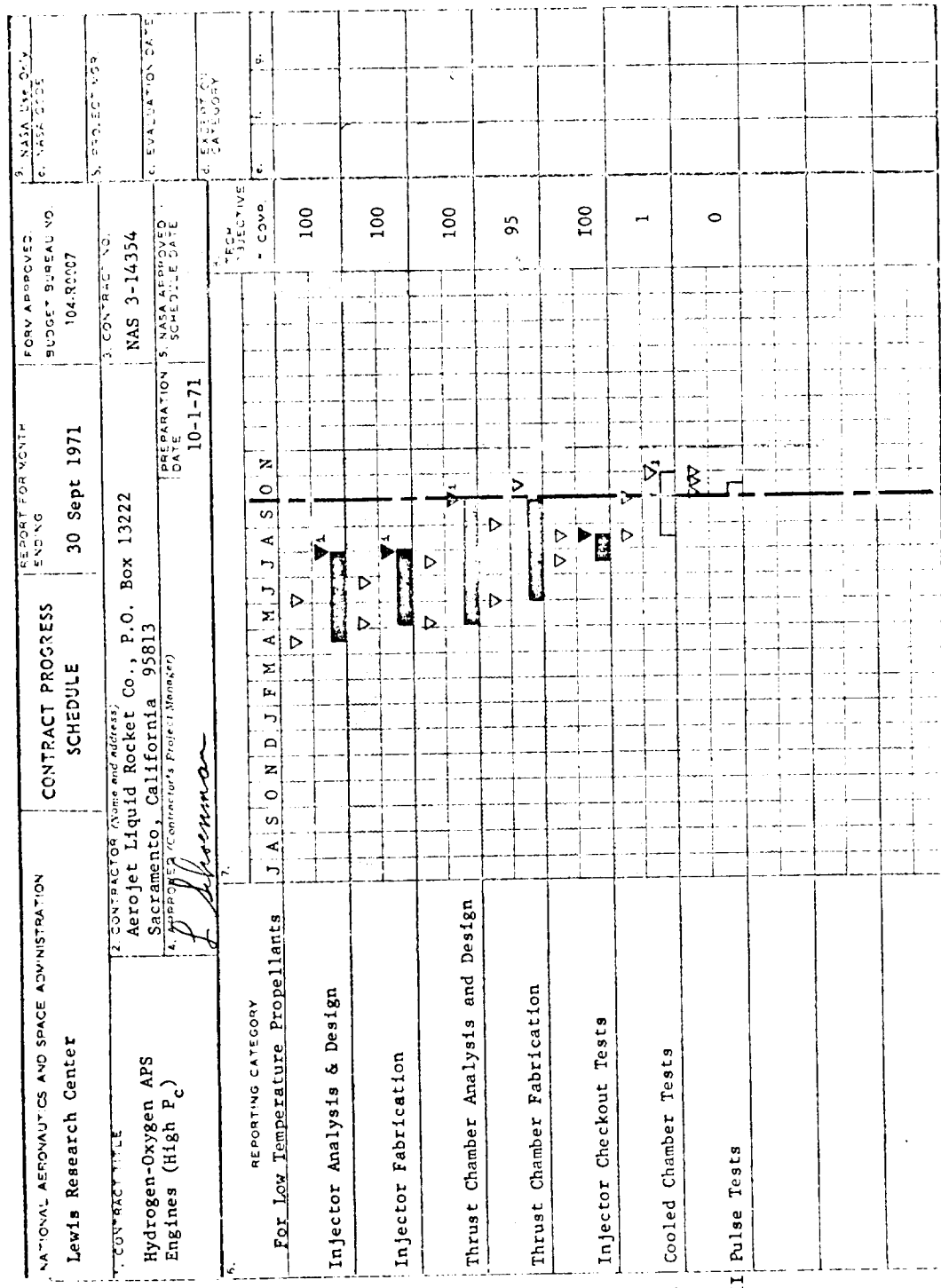
Task XXI

6. REPORTING CATEGORY	7. 1970 J A S O N D / 1971 J F M A M J J A S O N	8. TECH OBJECTIVE % COMP.	e.	f.	g.
Program Plan and Test Plans					
High P_c		85			
Low P_c		100			
Monthly Reports		91			
Quarterly Reports		100			
Final Report Draft		25			

NASA C-61 (Rev. 1-68)

NASA APPROVED SCHEDULE

CONTRACTOR'S WORKING SCHEDULE


NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center
CONTRACT PROGRESS SCHEDULE
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2. CONTRACTOR (Name and address) Aerojet Liquid Rocket Co., P.O. Box 13222 Sacramento, California 95813
3. CONTRACT NO. NAS 3-14354
4. PREPARED (Contractor's Project Manager)
PREPARATION DATE 10-1-71
5. NASA APPROVED SCHEDULE DATE
Hydrogen-Oxygen APS Engines (High Pc)
REPORTING CATEGORY For Low Temperature Propellants
J A S O N D J F M A M J J A S O N
% COMP.
Task
XXII Injector Analysis & Design 100
XXIII Injector Fabrication 100
XXIV Thrust Chamber Analysis and Design 100
XXV Thrust Chamber Fabrication 95
XXVI Injector Checkout Tests 100
XXVII Cooled Chamber Tests 1
XXVIII Pulse Tests 0

Appendix A

TASK X PULSE TESTING

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Test Summary	3
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Igniter Sequencing and Repeatability Tests	4
Igniter Main Stage Sequencing	5
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FIGURE LIST

Figure	
1	Flow Schematic for Pulse Tests
2	Anemometer Response and Calibration for Pulse Testing
3	Igniter Propellant Sequencing Tests
4	Igniter Lead - Main Stage Sequencing Tests
5	Ignition Delay with Main Stage Fuel Leading Igniter Ignition
6	Oxidizer Manifold Overpressure Experienced with Fuel Lead Sequence
7	Ignition Transient and Pulse Characteristics with Simultaneous Fuel and Oxidizer Flow
8	Pulsing Characteristics for Nominal Pulse Width of 0.2 sec
9	Pulsing Characteristics for Nominal Pulse Width of 0.1 sec
10	Pulsing Characteristics for Nominal Pulse Width of 0.035 sec
11	Film Cooled Chamber Temperatures in Pulse Mode Operation, 110 Pulses, Pulse Width 0.2 sec
12	Thruster Temperatures in Pulse Mode Operation, 1220 Pulses, Pulse Width .035 and .010 sec

TEST OBJECTIVES

The premix "I" triplet injector, film cooled thrust chamber and 25-lbf spark igniter were selected for pulse testing based on the high performance and excellant durability exhibited in the long duration firings of Tasks IX and XXVI. The objectives of this test series were to:

1) Investigate the start transient dynamics of a soft system (no mass flow controller),

2) Investigate thermal characteristics and heat soaks under various duty cycle modes,

3) Investigate bit impulse repeatability for values down to 50-lbf-sec and bit specific impulse for various pulse widths, and

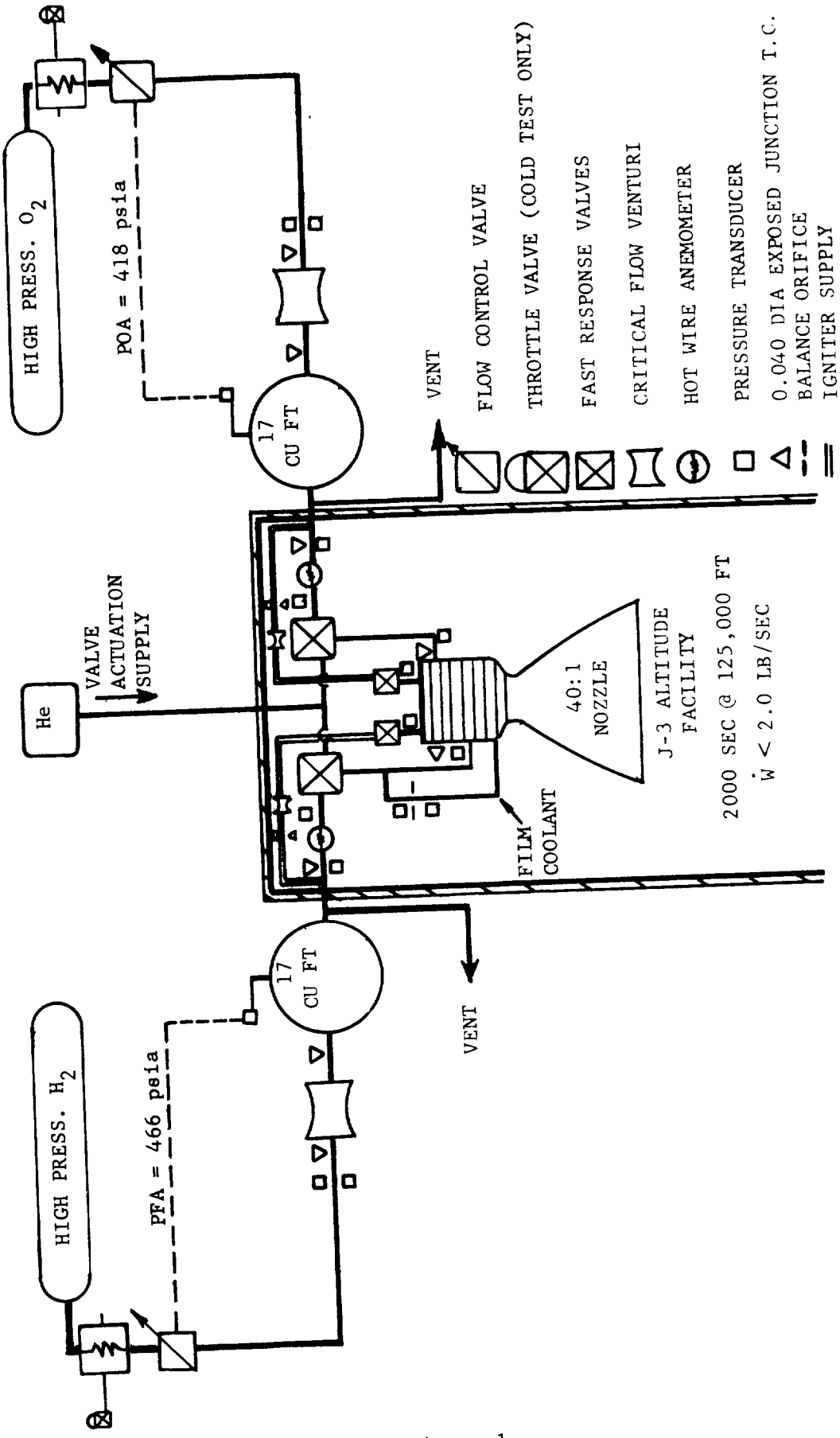
4) Demonstrate thruster durability in a series of firings involving 2500 pulses.

SYSTEM

A simplified schematic drawing of the J-3 Altitude Test Facility as configured for the pulse test series is shown in Figure 1. The features of this test setup are as follows:

1) One fuel and one oxidizer accumulator tank (17 cu ft each) located 23 feet upstream of the thrust chamber valves are the source of all engine propellants including those flowing to the igniter.

2) Oxidizer and fuel accumulator tanks are maintained at pressures of 430 and 466 psia in order to provide nominal thrust and chamber pressure of 1500-lbf, 300 psia, at a mixture ratio of 4.0.



FLOW SCHEMATIC FOR PULSE TESTS

Figure 1

System (cont.)

3) Flow to the igniter is regulated by high recovery venturis which flow choked during the igniter start transient.

4) Fuel and oxidizer flows are measured by hot wire anemometers located 10.5 in. upstream of the thrust chamber valves, with redundant measurements provided by critical flow control orifices located upstream of the accumulator tanks.

5) Thrust on the firing fixture having a 100 cps natural frequency is measured via a dual bridge 5000-lbf load cell.

The duty cycle and valve sequencing for pulse mode operations are established by a preprogrammed analog computer located in the test area control room. The computer regulates throttling valves which maintain accumulator pressure via a pressure feedback loop. The analog computer is also programmed to monitor critical engine parameters and terminate a pulse train should a malfunction occur. Potential failure modes and parameters monitored to detect such failures are provided in the following table.

Potential Malfunction Mode	Parameter Monitored and Criteria for Continued Firings
Failure of a thrust chamber valve to open or failure to achieve ignition	Thrust must be greater than 750-lbf at .075 sec after first electrical signal.
Shift in valve response or actuation resulting in overpressures or hard starts.	Injector manifold pressures must be less than 400 psia at all times.
Component overheating.	Injector face, throat, and skirt temp. less than 1000, 1500, and 2000°F, respectively.

HARDWARE & INSTRUMENTATION

Test hardware is described under Task IV chamber design; instrumentation is discussed under Tasks IX and XXVI film cooled chamber testing. Three data acquisition systems were employed to record thrust, flow rates, temperature and pressure.

1) Millisadic (digital recorder) 500 samples per sec per channel
2) High frequency FM tape
3) Oscillograph

TEST SUMMARY

Testing was divided into four categories:

1) Hot wire anemometer flow calibration
2) Igniter sequencing and repeatability tests
3) Igniter-thruster sequencing tests
4) Pulse performance and durability tests

CALIBRATION TESTS

Test 001 was an oxidizer anemometer flow calibration test in which three transducers were successfully calibrated in oxygen. The calibrations were accomplished in the test circuit with each anemometer in the correct run position for pulse flow. A downstream orifice was employed to simulate the thruster. The calibrations were made with ambient temperature propellants in a five-step flow range for the purpose of linearizing the transducer output. The flow measuring standard in these runs were the critical flow nozzles positioned upstream of the anemometers. Approximately 20 flow points were made on each anemometer. A typical analog output curve obtained during earlier calibrations can be found in Figure 2, which shows excellent full-scale response of approximately 10 millisec and a minimum overshoot. It is felt that this curve represents the

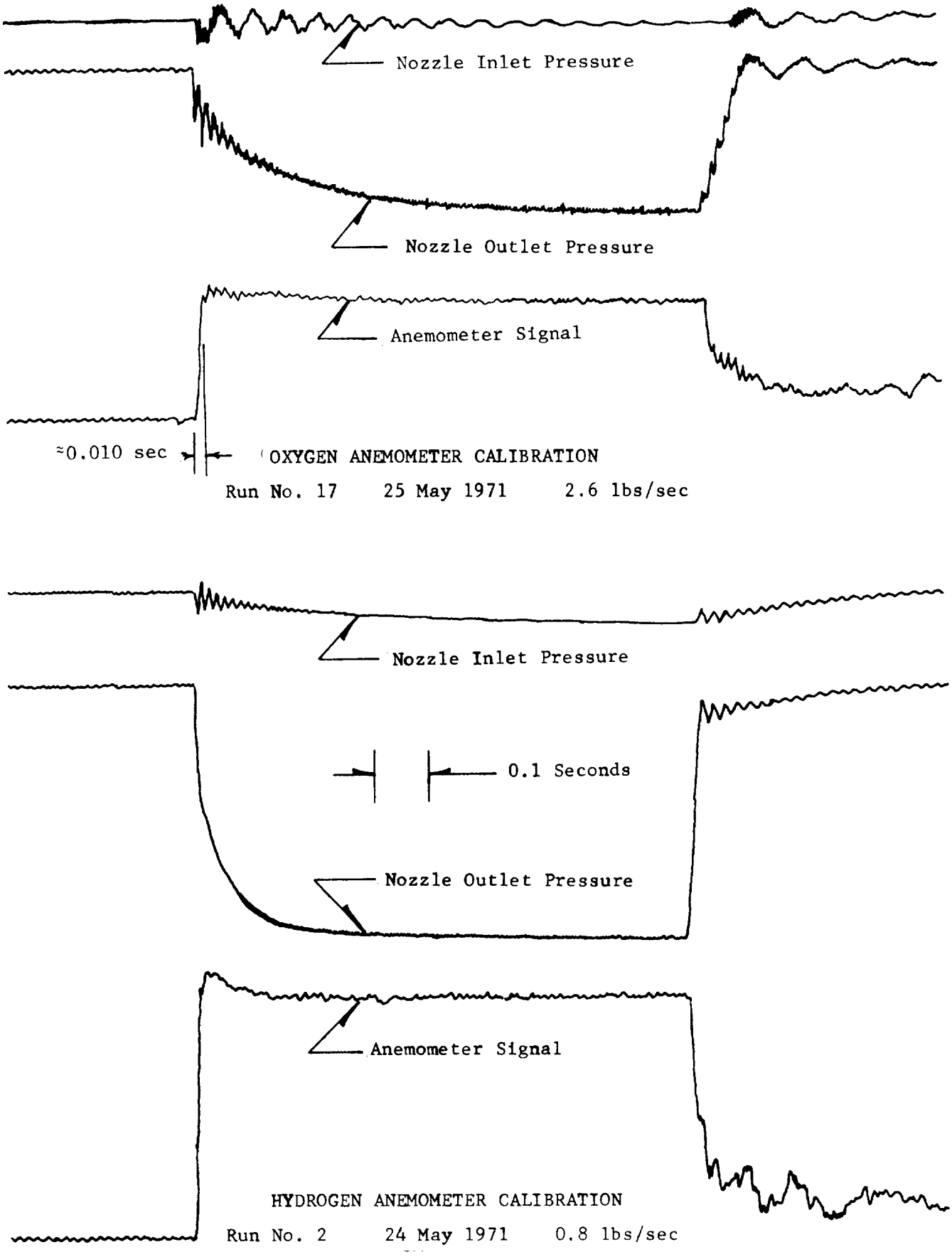


Figure 2 -- Anemometer Response and Calibration for Pulse Testing (Row Signal)

Calibration Tests (cont.)

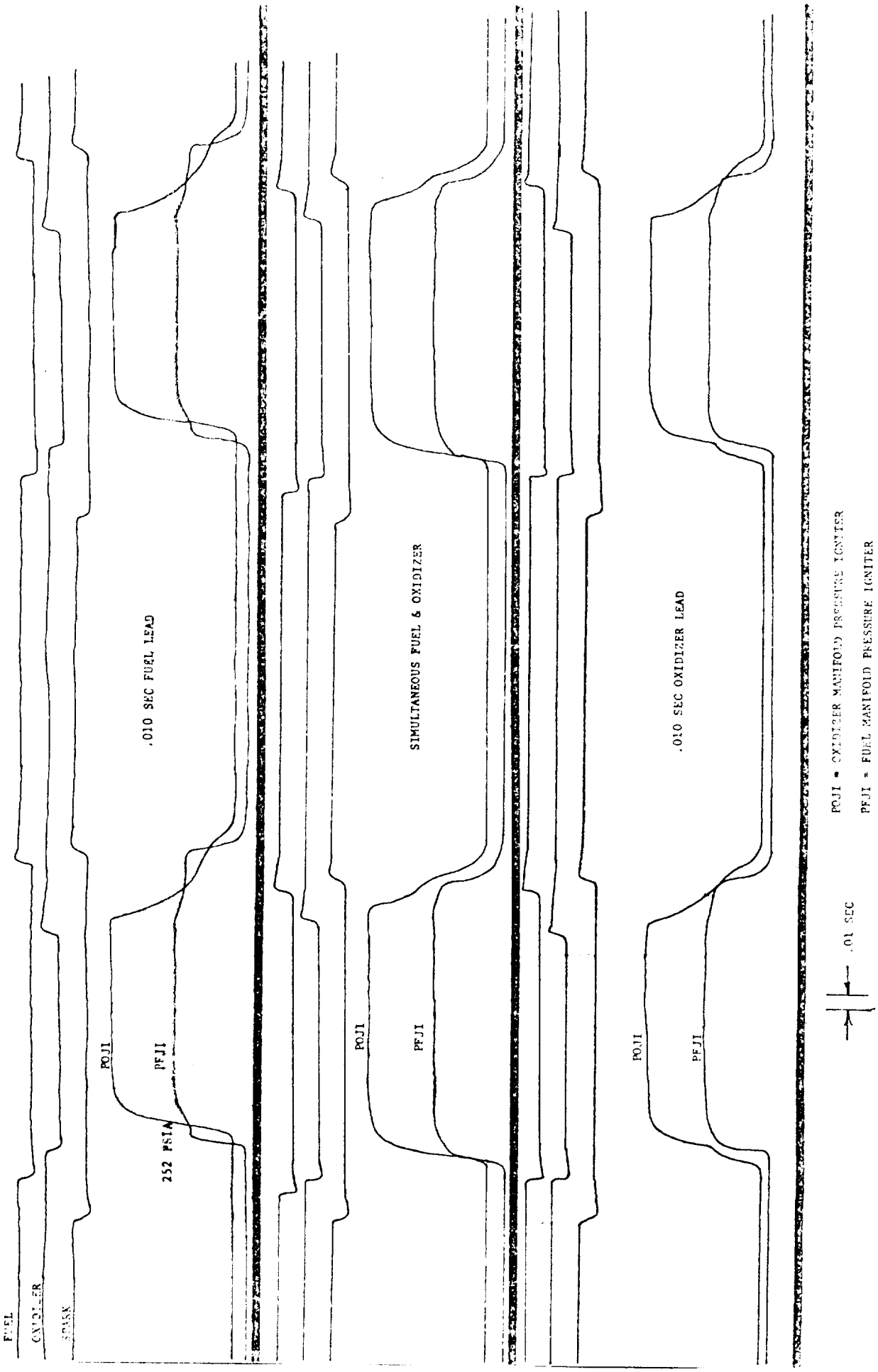
actual flow transient in the system during the 10 millisec valve opening and closing for the system in which the mass flow control venturi is employed. It will later be seen that a significant flow overshoot exists in a soft system in which the mass flow control device is removed.

Test 002 was a comparable test in which four fuel anemometers were successfully calibrated in the correct run position.

IGNITER SEQUENCING AND REPEATABILITY TESTS

Test 003 was an igniter only pulse series test in which fuel and oxidizer lead sequences were investigated. The conditions investigated were .010 sec fuel lead, simultaneous pressurization of fuel and oxidizer manifolds, and .010 sec oxidizer lead. A significant number of 1, 10, and 20 pulse trains in the igniter only mode were run during this period. Data acquisition was in the form of oscillograph traces, in which only the longer 20-pulse sequences were given run numbers, 003 a, b, and c. Typical results of these tests are shown in Figure 3, which depicts the igniter start transients with various valve sequencings. This test series demonstrated that it is possible to achieve satisfactory igniter ignition with a variety of valve sequences. A .010 sec fuel lead was selected for the pulse test series based on a vast backlog of experiences indicating that this sequence is least sensitive to potential high voltage electrical shorts. It also requires the lowest spark power level. Spark gaps of .025 and .050 were briefly evaluated during this period and the smaller gap was found to provide greater reliability because it insures that the spark discharge does take place at the spark gap.

During the course of these multiple igniter only mode firings at the nominal 25-lbf, MR = 6 for .20 sec (5-lbf sec impulse), the tip of the copper film cooling injection ring overheated and flowed slightly. The damage was at this point apparently so minor that it was not detected in normal hardware inspection. However, based on these test results and hardware damage, igniter only mode operation must be considered limited. Several pulses per minute for



Igniter Propellant Sequencing Tests

Figure 3

Igniter Sequencing and Repeatability Tests (cont.)

short periods are probably acceptable, while the two pulse per second rate evaluated is not.

IGNITER MAIN STAGE SEQUENCING

The next series of tests investigated the relationship between igniter ignition and main stage valve sequencing, in an attempt to minimize the time from first electrical signal to full thrust. Ignition delay times (lagging propellant flow to observable ignition) are found to be in the order of one or two thousandths of a second. With a fuel lead on the igniter, igniter ignition occurs as soon as the oxidizer reaches the spark gap. Main stage ignition, with the igniter running, can be detected within .005 sec after initial movement of the lagging valve. Most of this time is probably fill times and delays in sensing ignition due to transducer line lengths.

Four values of igniter leads (igniter ignition to TC oxidizer valve movement "LOV" were investigated in this series, three of which are shown in Figure 4. The normal ignition sequence (left), Run 04-012, employed in the previous long duration firings provides a .050 sec igniter lead with the igniter oxidizer sequenced off .04 sec after main stage ignition. Experience on hundreds Task VIII, IX, and XVI firings demonstrated this to be a 100% reliable sequence. The first reduction in this igniter lead period investigated in Task X test 004 was a .020 sec igniter lead. Satisfactory ignition was achieved. Test 005 was a repeat with comparable results. In test 006 two sequencing change errors were made which (1) reverted the igniter to .002 sec oxidizer lead and (2) gave a .02 sec delay between main fuel and oxidizer valve opening (oxidizer lag). This sequence resulted in failure of the igniter to ignite and there was a low thrust shutdown. Failure appears to be due to a lack of spark at the spark gap. Test 007 corrected the sequence error and reduced the igniter lead to .012 sec. Satisfactory ignition was achieved. Tests 008 through 012 (see Report Table I) were

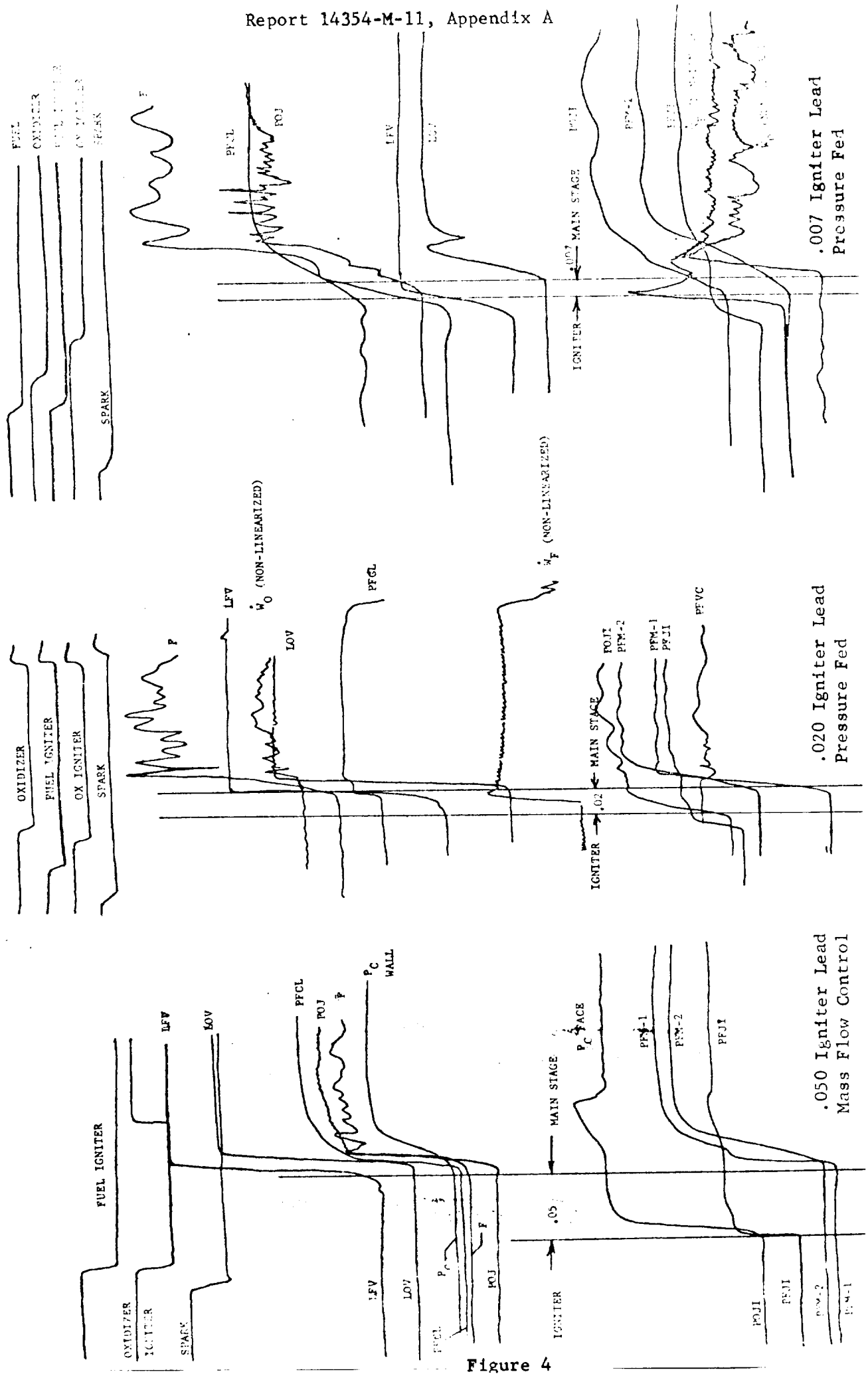


Figure 4

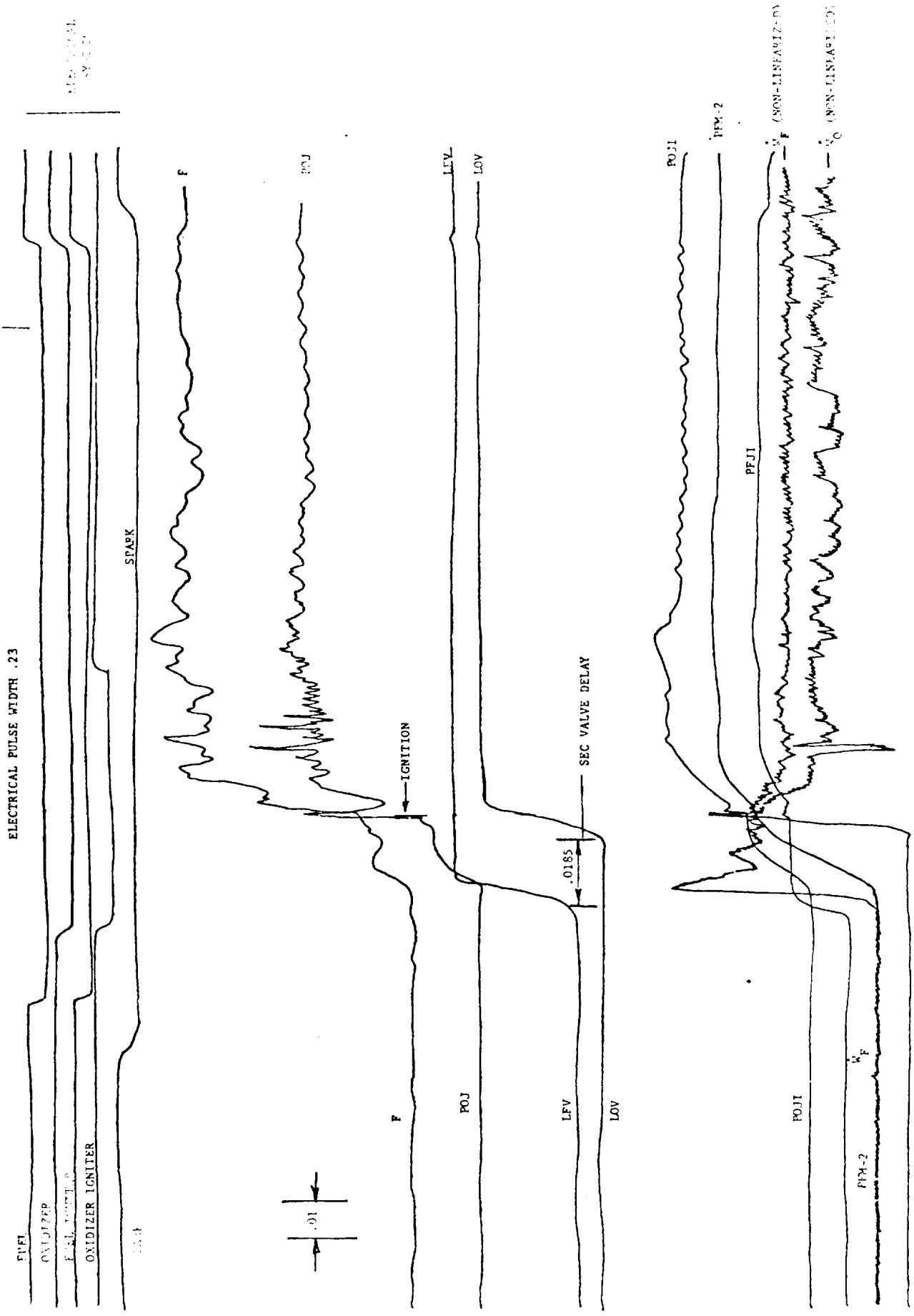
Igniter Lead - Main Stage Sequencing Tests

Igniter Main Stage Sequencing (cont.)

final sequence adjustments to further reduce the igniter leads, to look at multiple pulses and investigate variable off times. Coast time between pulses were started at 10 minutes and reduced to 0.3 sec. Two and five pulse train series were conducted. The final igniter sequencing used in the pulse series, shown on the right of Figure 4, was a .007 sec lead to LOV start. A difference in the thrust and P_c rise rates, thrust overshoots and general flow characteristics is noted between the .050 sec igniter lead and other two sequences shown. This is due to the use of a mass flow controlled feed system in the long duration tests (left) and a soft or nonflow controlled supply in the fixed pressure tank system shown center and right. The thrust rise rate (0 to max. thrust) is slightly faster in the soft system for the flow configuration tested.

The igniter only mode damage to the tip of the film cooling ring was descovered during a normal hardware inspection following test 012. The damaged area was removed by machining approximately .080 in. off the tip, and normal testing was resumed.

Test 013 was a .2 sec pulse sequence check on the reassembled hardware with repeatable satisfactory ignition. In test 014 the fuel valve was advanced an additional .01 sec (.02 from LFV to LOV) to determine the effect of a significant fuel lead. In this sequence the main fuel valve was sequence opened prior to flow of the igniter oxidizer, i.e., .005 sec igniter lag. This resulted in a back pressure within the entire thruster. The injector and igniter were filled with H_2 when the oxidizer igniter valve opened, thus increasing the time for the oxidizer to reach the spark gap. This resulted in an igniter ignition delay of .02 sec. Igniter ignition occurred about the time of the opening of the main stage oxidizer valve, and there was a very rapid rise in POJ but no over-pressure. This sequence is shown in Figure 5.



Ignition Delay with Main Stage Fuel Leading Igniter Ignition
(1680-D05-0A-014)

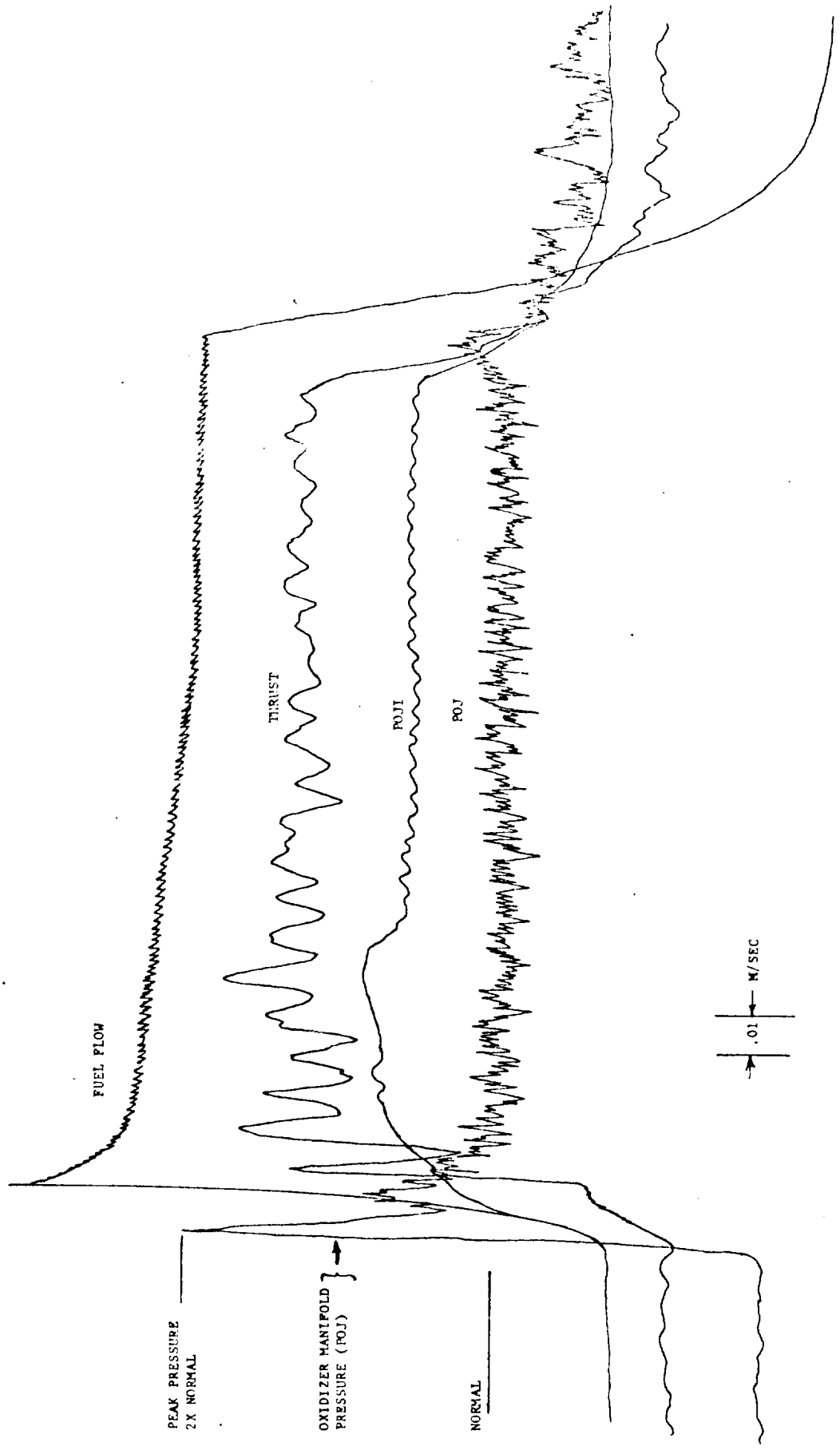
Figure 5

Igniter Main Stage Sequencing (cont.)

Test 015 reverted to the normal sequence, and 100 pulses were executed "all alike". Test 016 provided an additional 110 pulses. The computer terminated the pulse train on the 110 pulse due to low thrust. Normal ignition was attained but oxidizer flow was low. Tests 017 and 018 investigated the reason for the low flow and it was determined that there was flashback and overpressure in the oxidizer manifold starting at around the 100th pulse. This deformed a sheet metal diffuser plate in the manifold obstructing the oxidizer orifices. Oxidizer manifold pressures (POJ) of approximately 800 psia (100% overpressure) were recorded on high frequence FM tape for several pulses prior to this failure. Figure 6 shows a playback of this tape for one of the final pulses. The overpressures appear to be the result of a backflow of fuel into the oxidizer manifold during the start sequence when operating with the fuel lead. For a short time after the oxygen valve is opened, a combustible mixture exists in the oxidizer manifold, giving rise to the flashback condition. There appears to be no reason a similar situation could not be encountered in the fuel manifold given a sufficient oxidizer lead. This problem is probably characteristic of all gaseous propellant thrusters and is best solved either by closely controlling sequencing or else simply making the manifold structurally capable of withstanding the overpressure.

Following the flashback, the SN-7 injector was placed on the stand and the sequence changed to provide a slight oxidizer lead on injector manifold fill and the pulse counter was reset from "220" back to "0" so that a full 2500 pulses would be demonstrated on a single injector. Figure 7 provides for comparison, the main stage ignition employing simultaneous main stage valve openings as recorded on the high frequency FM tape.

The pulsing was resumed with Test 019 and continued through 021 during which 269 identical 0.2-sec pulses were accumulated. A "no light" condition was encountered on the 266th pulse of Run 021 which caused a low thrust shutdown. This "no light" was attributed to a failure of the igniter power supply. While



Oxidizer Manifold Overpressure Experienced with Fuel Load Sequence

Figure 6

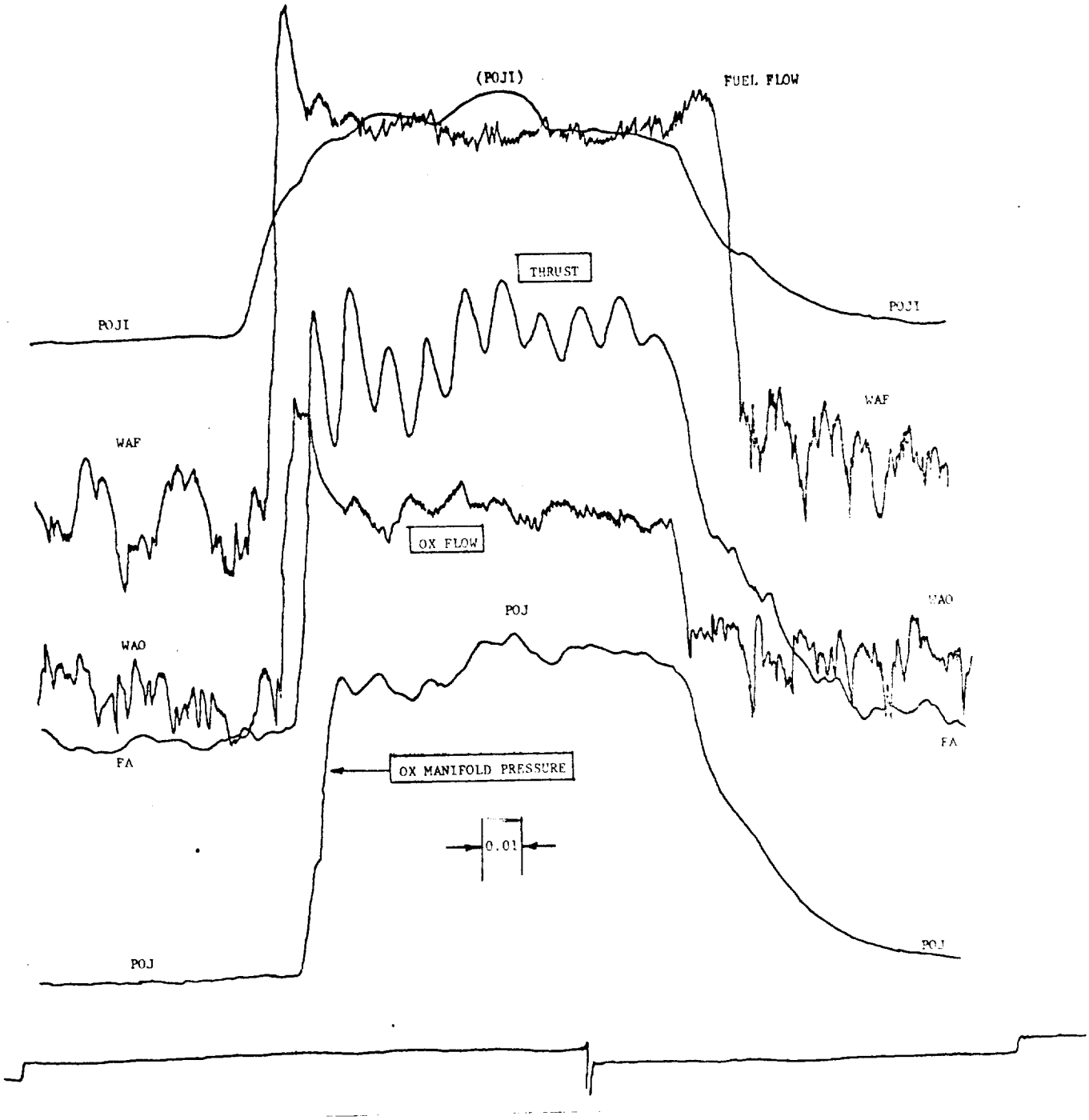


Figure 7 -- Ignition Transient and Pulse Characteristics with Simultaneous Fuel and Oxidizer Flow

Igniter Main Stage Sequencing (cont.)

testing on Contract NAS 3-14348, it was found that, during repeated operation, a solenoid in the power supply box would overheat, producing intermittent sparking. Test 002 was a single pulse with high recording speed on the oscillograph to obtain better data resolution. Response and ignition were normal. Testing resumed with Run 023 for 259 pulses, all normal and repeatable. In test 024, the pulse width was reduced to 0.1 sec, which was continued through test 028, during which 770 additional pulses were executed without difficulty, all being repeatable. Test 028 involved 760 continuous pulses.

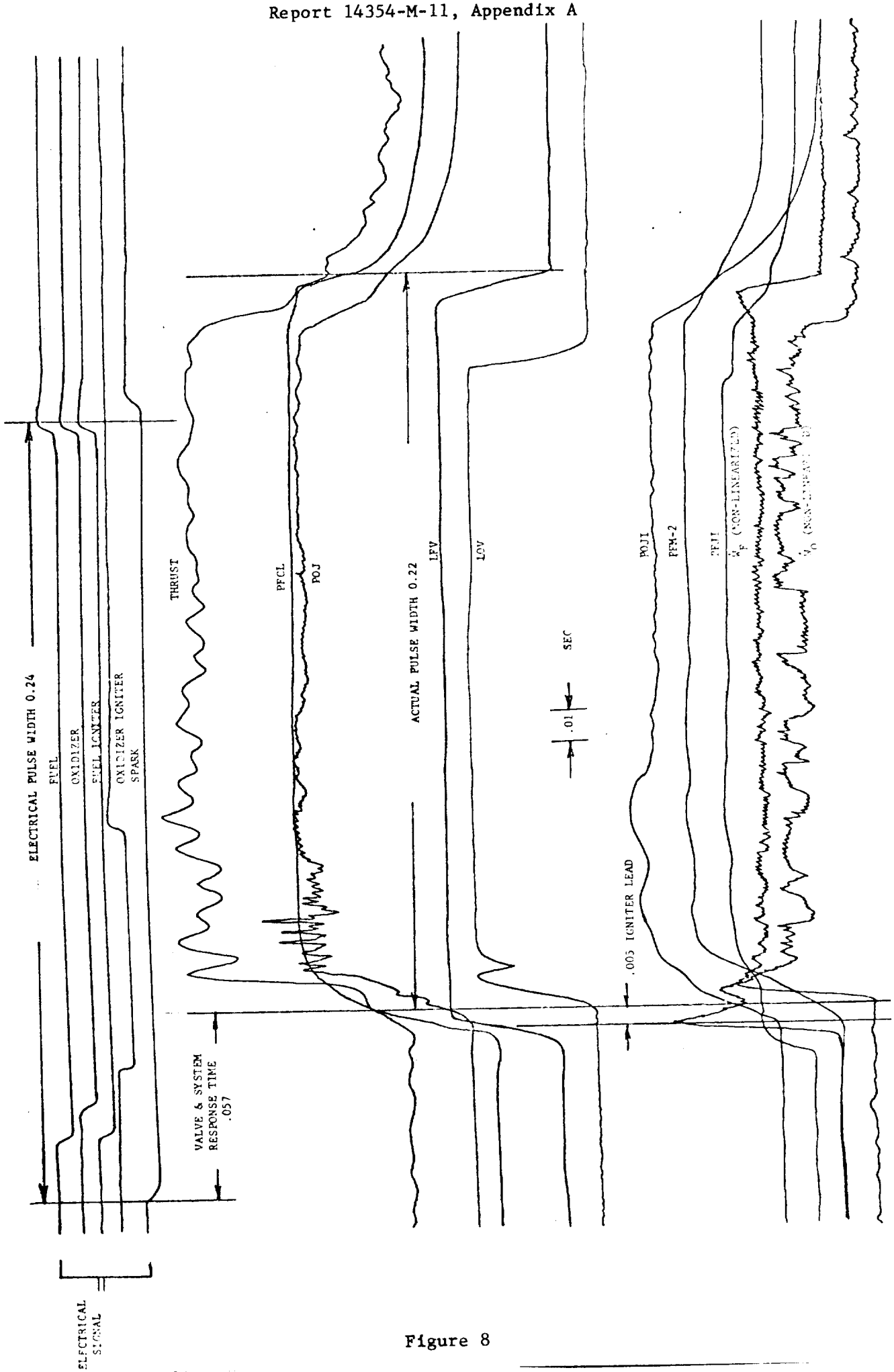
Tests 029 through 037 further reduced the pulse widths to .035 sec and added an additional 395 pulses. The main stage fuel valve operation became irregular after approximately 250 pulses in test 037 due to overheating of a pilot valve. At this point in time the thruster was executing 10 pulses per second which exceeded the cooling capability of the pilot valve electronics. Other than the valve action during the final few pulses, all pulses were repeatable.

In test 039 the duty cycle was returned to the .10 pulse width and 840 additional firings executed in rapid succession.

Representations of the oscillograph traces for typical .2, .1, .035 pulse widths are shown in Figures 8, 9, and 10.

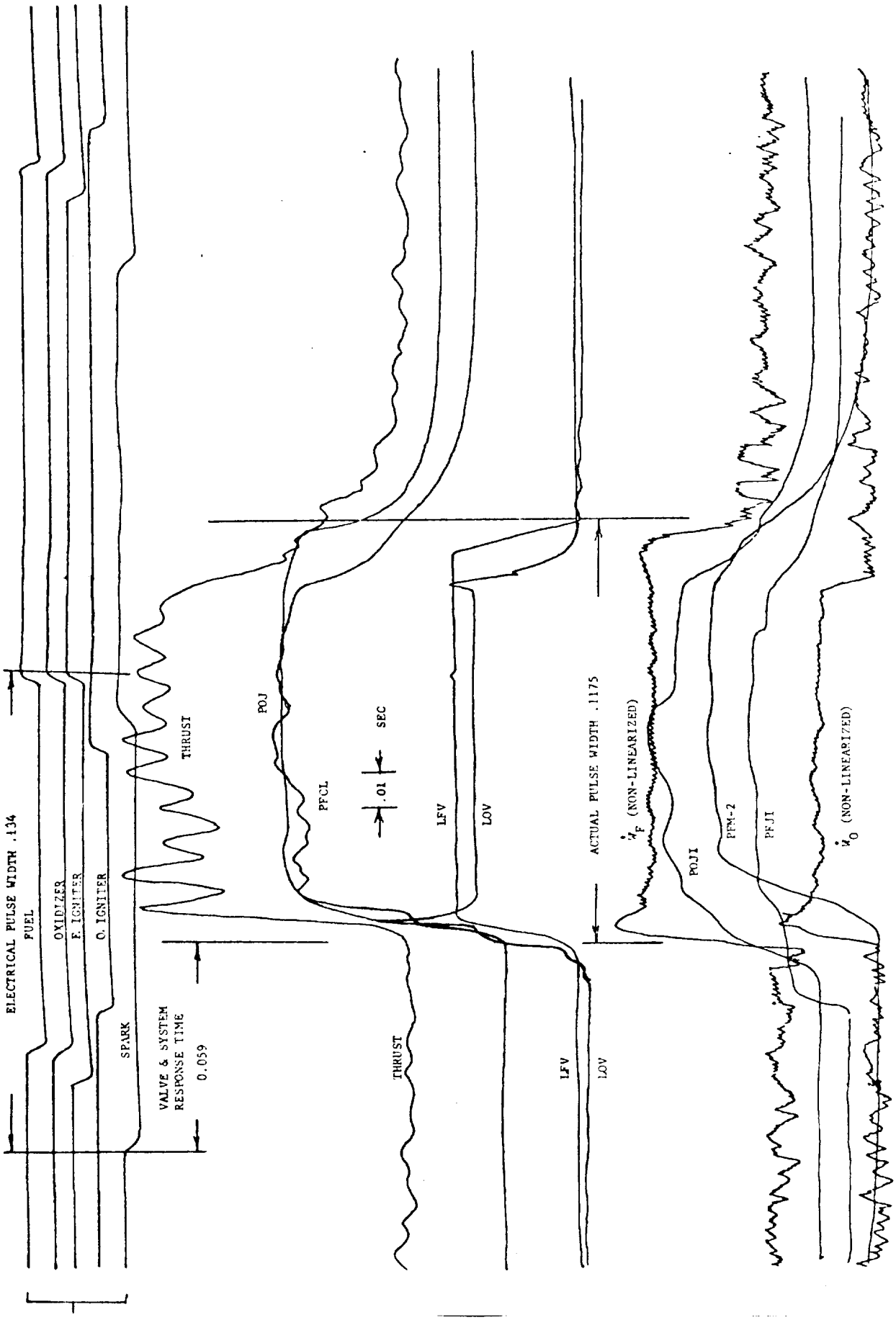
TEST RESULTS

No pulse mode limitations or heat soak problems were detected with 20% fuel film cooling. Operating temperatures are lower than steady-state firing. Temperatures recorded on control room Brown recorders during the pulse tests are shown in Figures 11 and 12. Performance parameters from these tests are currently being reduced and reviewed.



Pulsing Characteristics for Nominal Pulse Width of 0.2 sec
(1680-D05-0A-016)

Figure 8



Pulsing Characteristics for Nominal Pulse Width of 0.1 sec
(1680-D05-0A-028)

Figure 9

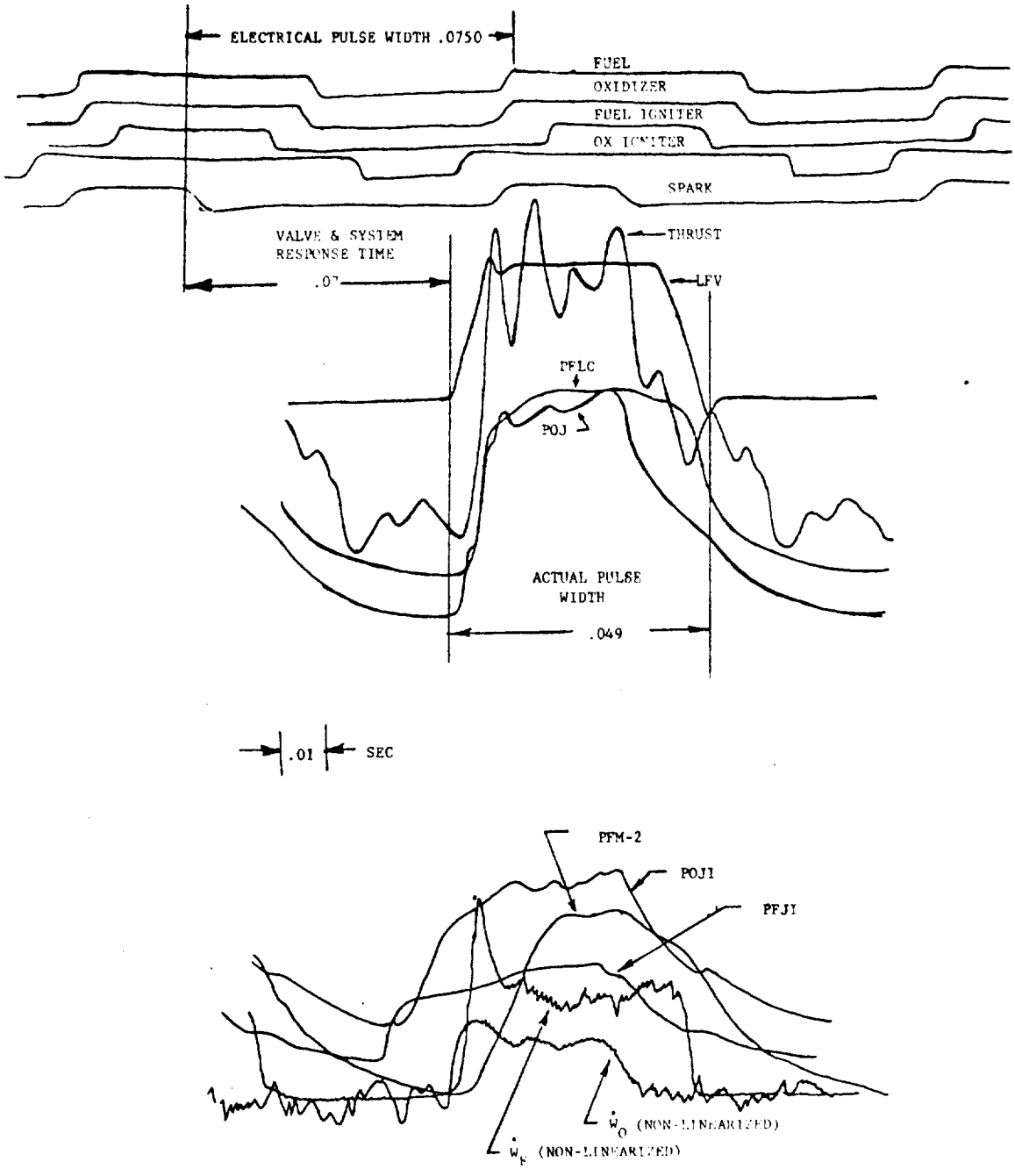


Figure 10 -- Pulsing Characteristics for Nominal Pulse Width of 0.035 sec (1680-D05-0A-036)

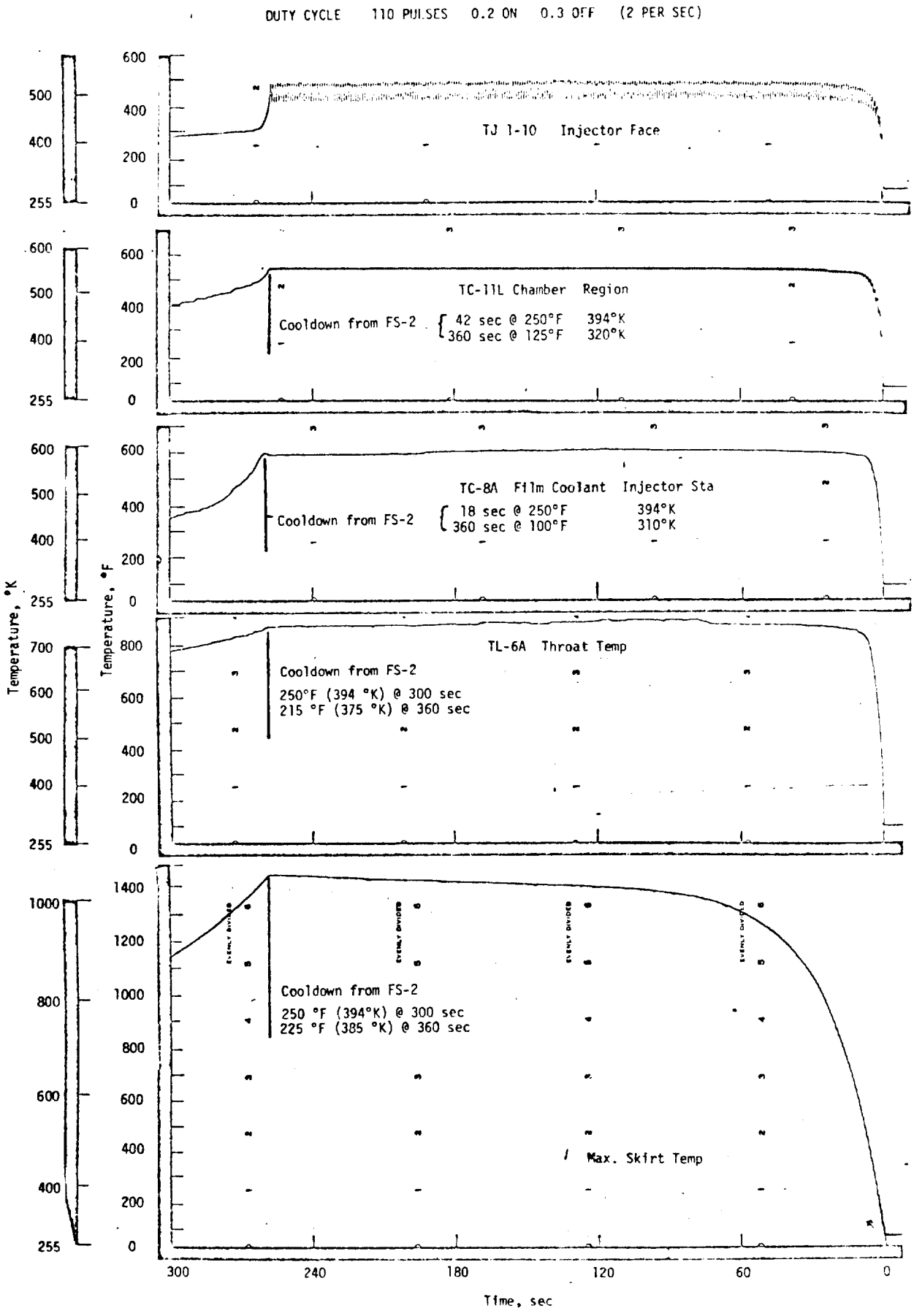
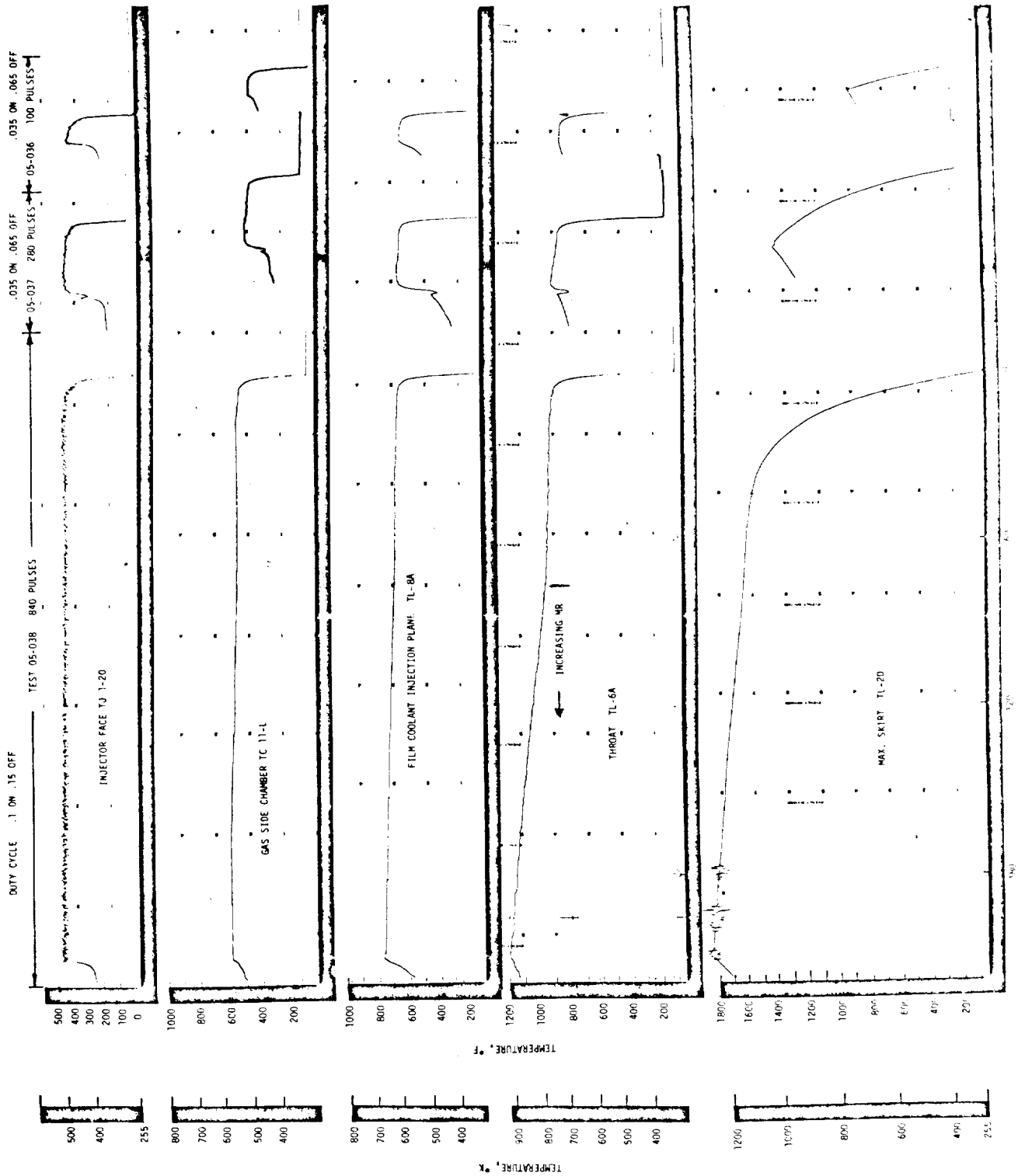


Figure 11 -- Film Cooled Chamber Temperatures in Pulse Mode Operation
110 Pulses, Pulse Width 0.2 Sec



Thruster Temperatures in Pulse Mode Operation
1220 Pulses, Pulse Width .035 and .010 Sec

Figure 12

Appendix B

DESIGN ANALYSIS FOR PROTOTYPE FILM COOLED CHAMBER WITH LOW TEMPERATURE PROPELLANTS

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1	Transient and Steady-State Temperature in Convectively Cooled Region
2	Monowall Manifold Steady-State Temperature
3	Non Contacting Manifold and Liner
4	Dual Wall Manifold Steady-State Temperatures
5	Film Cooled Chamber Structural Analysis (Steady-State)
6	Throat Region Fatigue Life

TABLE LIST

Table	
I	Summary of Film Cooled Chamber Design Analysis

Design analysis of the prototype film cooled chamber for 250°R fuel temperature is summarized in Table I. These include analysis of the short regeneratively cooled chamber region, the film coolant injection channels in the convergent nozzle, the film cooled throat and skirt, and the fuel inlet manifold region. The results of these analytical activities are presented and discussed with the aid of the figures in this section.

THERMAL

Transient and steady-state 2-dimensional heat conduction analyses were conducted on all chamber components with the exception of the manifold for which a 3-dimensional network was prepared and evaluated at steady-state thermal conditions. The thermal boundary condition employed were the recovery temperatures, gas and coolant side heat transfer coefficients derived from Task VIII heat sink, and Tasks IX and XXVI 40:1 highly instrumented cooled chamber tests. The gas side boundary condition inferred from the heat sink and cooled chamber tests were in agreement. Theoretical coolant side coefficients include adjustment factors for entrance effects and roughness of the small channels such that the predicted and measured wall temperatures are in good agreement.

Figure 1 shows the results of the transient analysis in the copper lined chamber region which includes a short regeneratively cooled section (top) feeding the injector and the short conical, convectively cooled, film cooling injector (bottom). The purpose of these analyses is to establish the temperature profiles for the thermal fatigue analysis and to determine the physical location and point in time at which maximum thermal strain occurs. Temperature-time plots for three axial chamber positions and four film coolant ring positions are shown. These are obtained from 2-dimensional models representing the wall cross-section.

From the analyses conducted to date, several general conclusions have been drawn regarding the time of maximum thermal strain. The time of maximum strain depends upon the (1) initial chamber temperature relative to the propellant temperature, (2) the wall thickness and nature of cooling channels, and (3) the wall material. The worst thermal condition will be at steady state if the wall

TABLE I

SUMMARY OF FILM COOLED CHAMBER DESIGN ANALYSES
(Nominal Operating Conditions)

Location	Analyses	Results
Chamber	Thermal - 2-dimensional transient and steady state, several channel geometries	Worst thermal condition at steady state; max. wall temp 320°F, max. ΔT 162°F 80 channels, .050 wide .150 deep .035 wall
	Structural - 2-dimensional finite element plastic at worst thermal conditions.	Max. strain .25 % steady state. Thermal η_f = 100,000
Film Coolant Injection Section	Thermal - 2-dimensional transient and steady state, several channel geometries	Worst thermal conditions .18 sec Trans. ΔT max = 200°F, SST max = 480°F
	Structural - 2-dimensional finite element plastic at .18 sec and thermal steady state	Max. strain .32% at .18 sec, $\eta_{f\ Therm}$ = 50,000 Steady state strain .26%, $\eta_{f\ thermal}$ = 90,000
Throat Region	Thermal parametric analyses of % FFC and wall thickness based on test data	Max. Temp at steady state < 1200°F Max Temp gradient at 0.05 to 0.1sec
	Structural 1 and 2-D plastic analyses transient and steady state (parametric results presented graphically)	10^6 thermal cycles 19% FFC .04 wall. Transient worst condition.
Skirt	Thermal analysis transient and steady state	Steady state worst thermal condition based on max temp 20% FFC 1500°F
	One and 2-D plastic structural analysis	Life not limited by high temp creep > 1000 hrs at 1800°F with stainless steel
Manifold	3-D thermal analysis of three designs. 2-D plastic analysis of two designs 2-D elastic analysis of one design	Results discussed in separate section. One design found superior in structural life. Manifold is not a limiting factor in life. Worst condition at steady state.

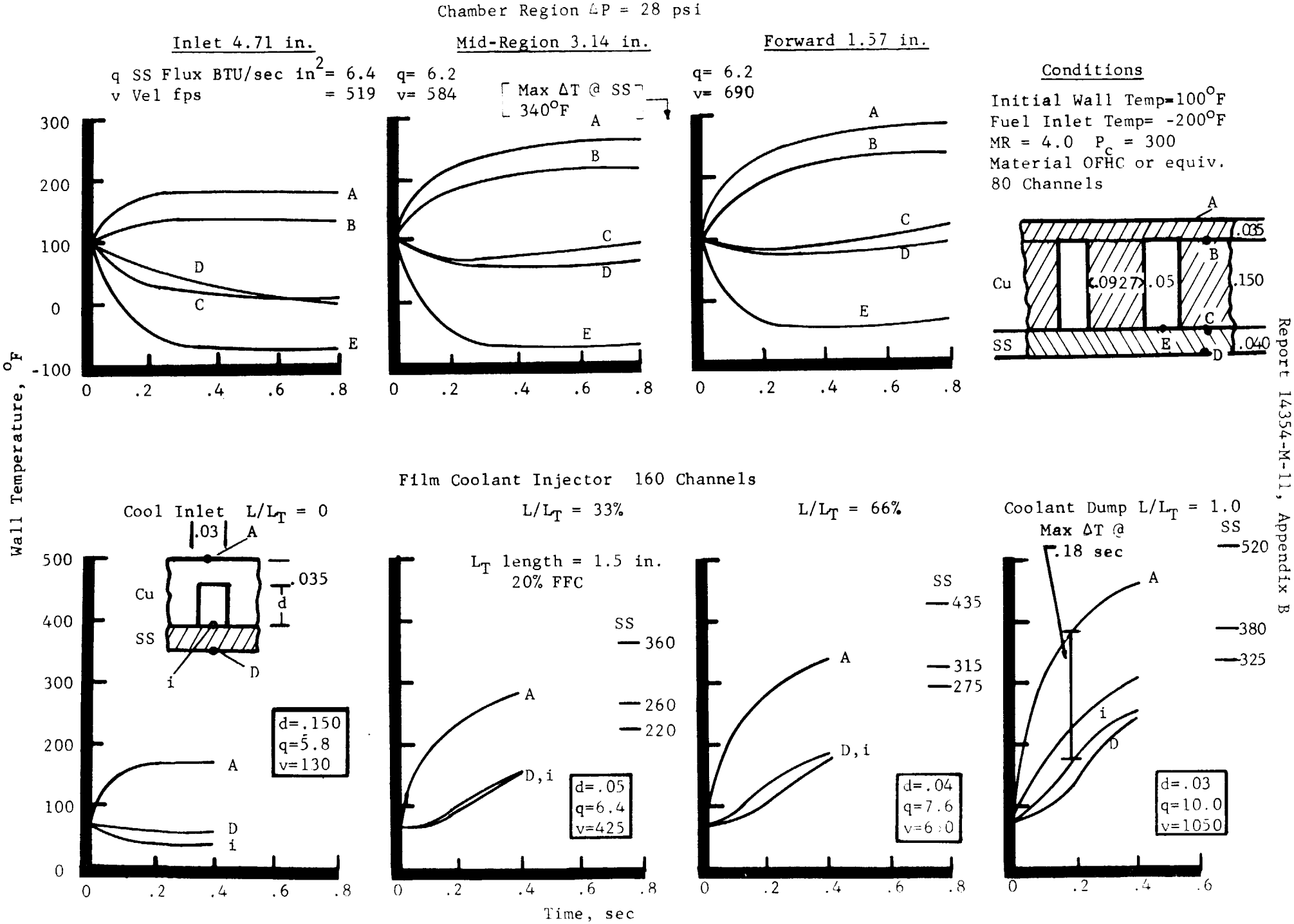


Figure 1

Transient and Steady-State Temperatures in Convectively Cooled Region

Thermal (cont.)

is relatively thick (large channel depth) and the initial chamber temperature is higher than the propellant temperature (530-560^{o}R soak temperature with 200-300^{o}R fuel coolant temperature). In this case firing the thruster results in cooling the backside while the gas side is heated. This is the situation in the chamber region.

The worst thermal strain will be found during the transient when both the gas and backside heat when the thruster is fired, either because the coolant and initial structure are at the same temperature, or because the wall thickness, material and coolant channel configuration allow penetration of a significant amount of heat to the exterior surface causing its temperature to rise. This latter condition exists at the discharge end of the film cooling injection channels, $L/L_T = 1$.

Examples of these results are shown in Figure 1, for nominal initial temperature conditions. In the chamber region, maximum thermal strain is found in the mid-region at steady state, while in the thinner conical film cooling region, maximum thermal strain occurs at the tip of the coolant injector at 0.18 sec into the firing.

A steady-state thermal map of the more highly stressed chamber and throat region including the supply manifold, is presented in Figure 2. This figure shows the results of the 3-dimensional steady-state thermal analysis which includes axial heat conduction as well as heat flow through the flow and peripherally around the cooling channels. In this activity emphasis was placed on providing an accurate description of the temperature gradients around the inlet manifolds. Since the exterior of the manifold will always cool down when the engine is fired, the worst thermal strain at steady-state rule applies. Three manifold configurations were investigated in this program. One was a single wall unitized design in which the manifold is made complient such that it can accommodate the expansion of the heated copper liner. A second approach

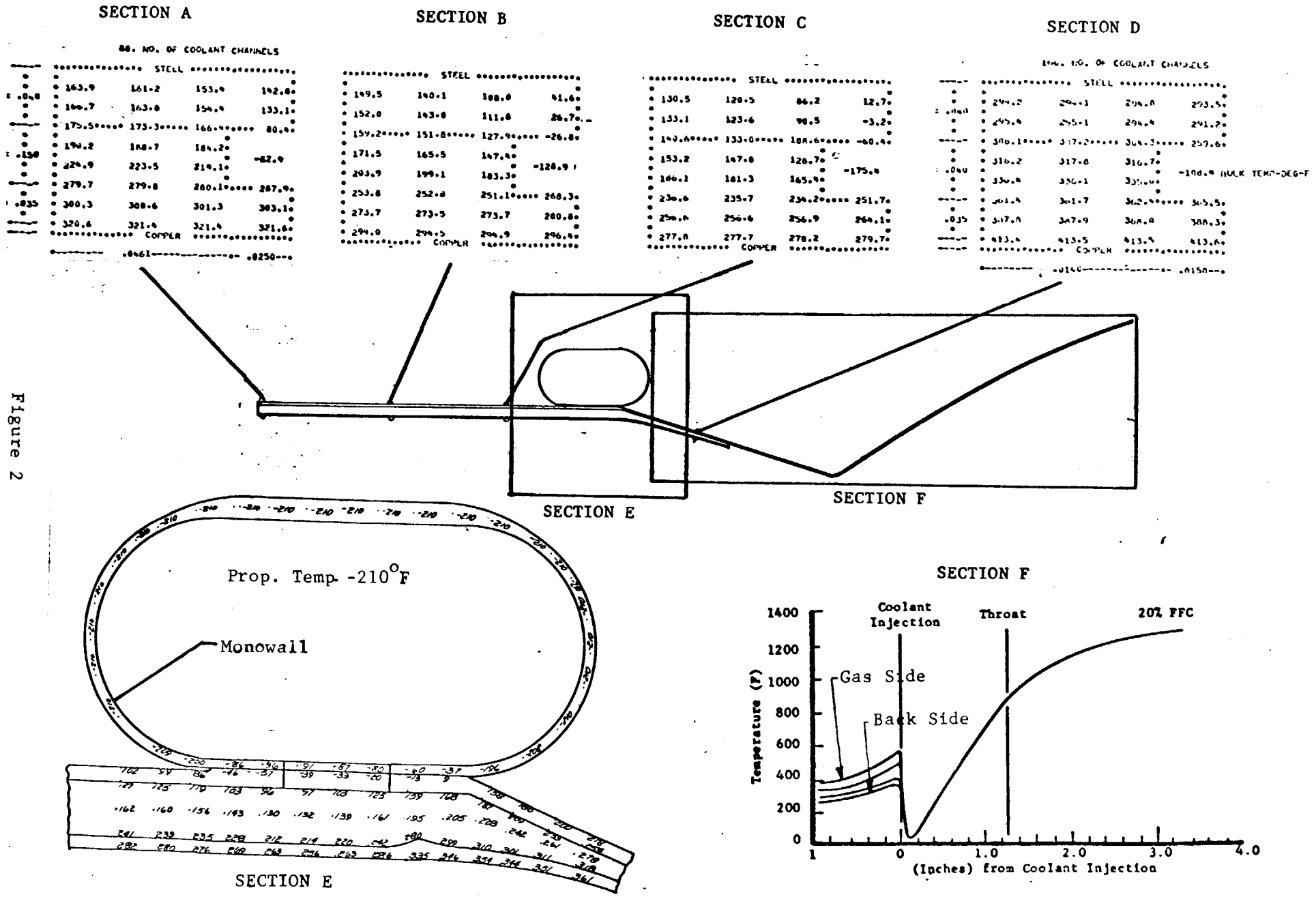


Figure 2

Monowall Manifold Steady-State Temperatures

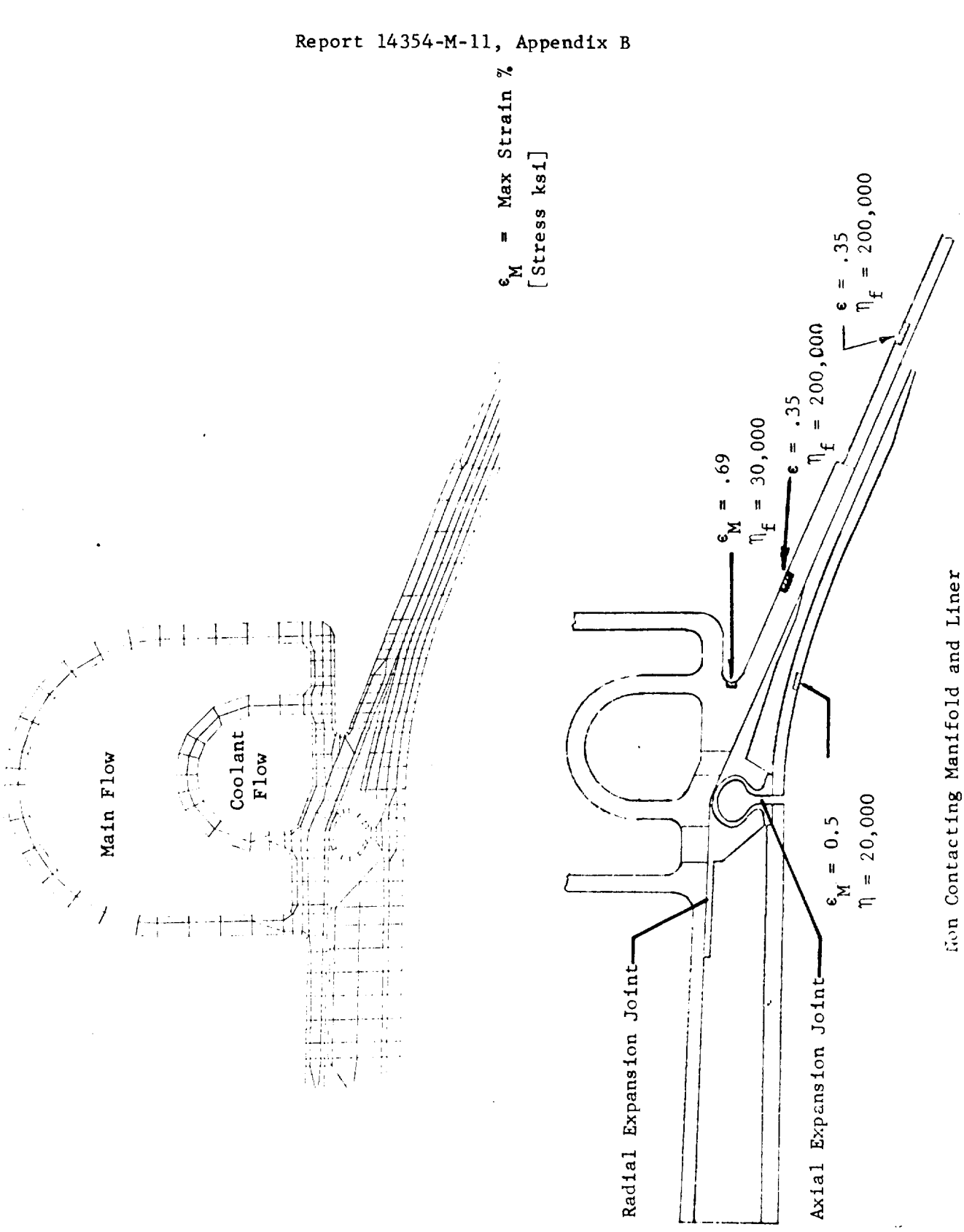
Thermal (cont.)

shown in Figure 3, had two ridged concentric manifolds, one each for main and coolant flow which allowed internal flow balancing. This approach accommodated thermal expansion via an axial expansion joint and radial, liner-to-jacket gap. A third approach was considered following structural analysis which showed several highly strained regions due to the large temperature gradients between the cold manifolds and the points where they attach to the chamber. The modified design incorporates a double complient wall manifold such that the cold H_2 flowing within the manifold contacts and cools only the inner wall, while the outer wall is attached rigidly to the structure. The outer manifold is the pressure structure and also supports the inner liner through a series of stand-offs, and local welds similar to the principle employed in most thermally isolated containers. Comparison of Section E of Figures 2 and 4, show that much smaller thermal gradients are found in the manifold attachment area of the dual wall design. The average manifold temperature is also much closer to the chamber skin temperature in the dual wall design due to the isolation from the cold H_2.

STRUCTURAL

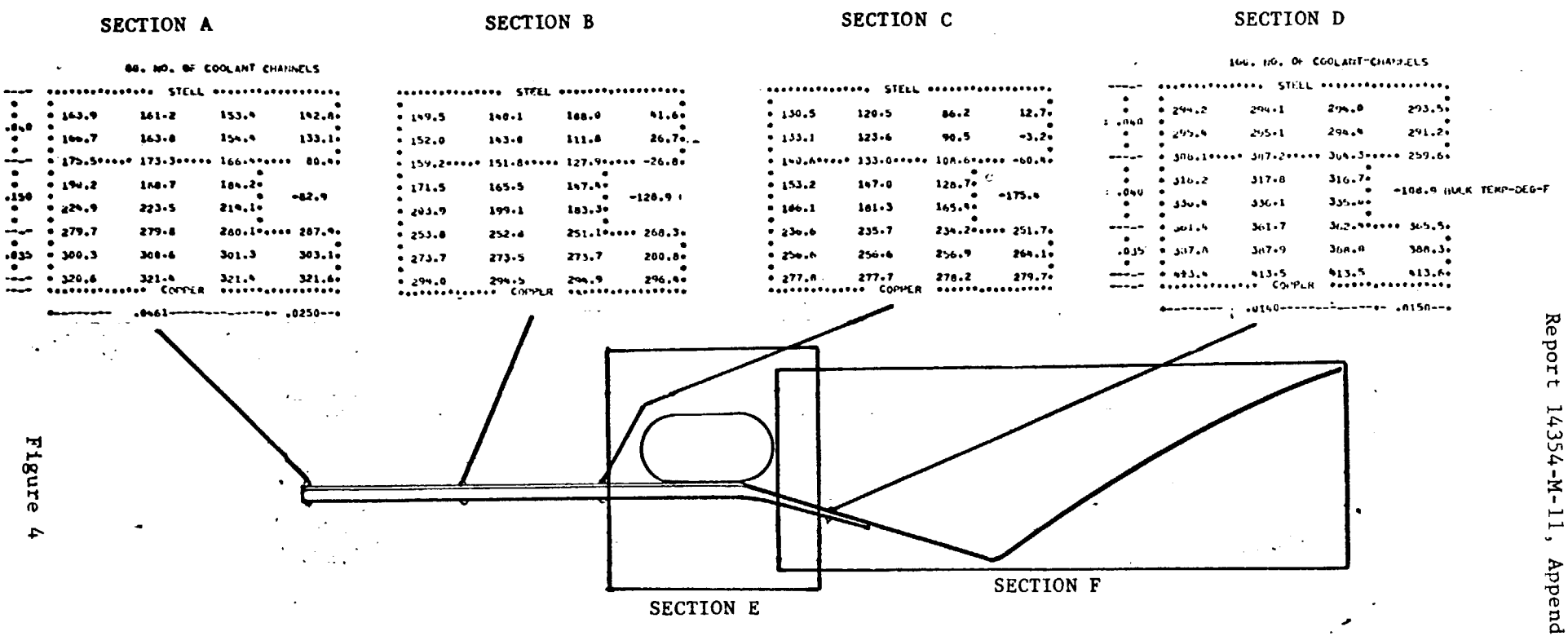
The finite element grids and the results of plastic structural analyses are shown in Figure 5. Manifold region grid and structural analyses are based on nominal operating conditions using the steady-state thermal maps shown in previous figures. The grid on the right, depicting the film cooling injection channels and throat region and grids similar to that shown lower left depicting the wall cross-section, were employed to investigate thermal strains during transient heating conditions.

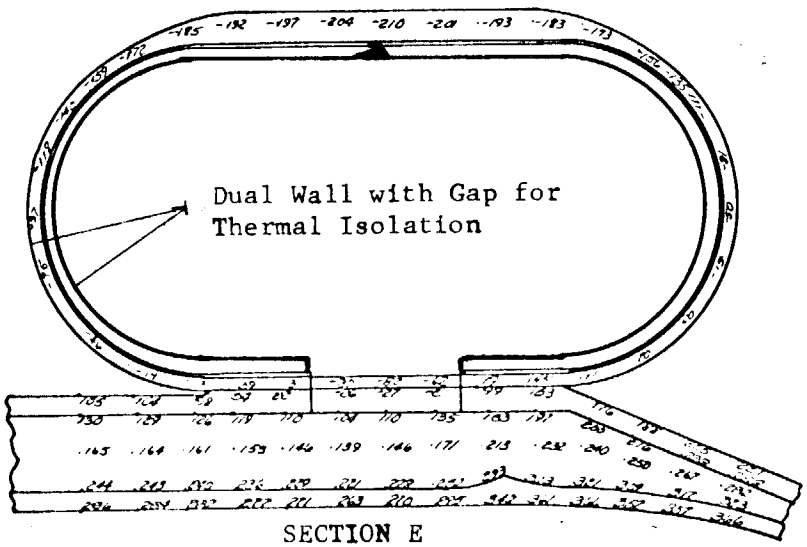
The recommended design is capable of providing 50,000 full thermal cycles with the limiting area being the tip of the film cooling ring. The chamber and manifold region is predicted to be useable for 100,000 thermal cycles and the throat region for 800,000 thermal cycles. The entire structure is adequate for 10^6 pressure reversal cycles with overpressure up to 850 psia without structural failure of any component.



Non Contacting Manifold and Liner
(Elastic Analysis)

Figure 3





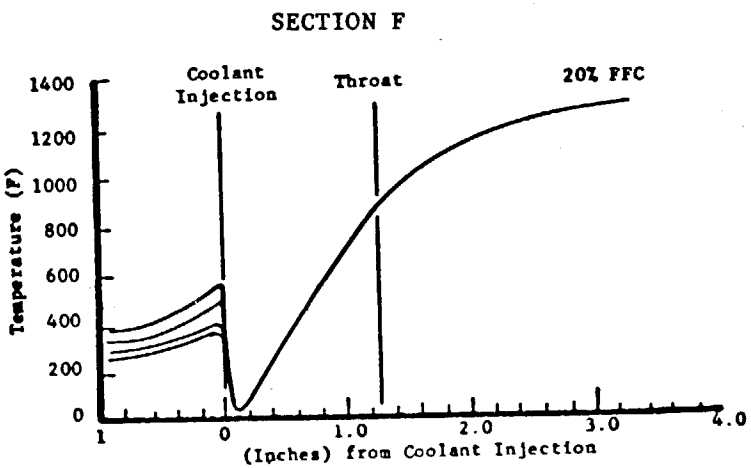


Figure 4

Dual Wall Manifold Steady-State Temperatures

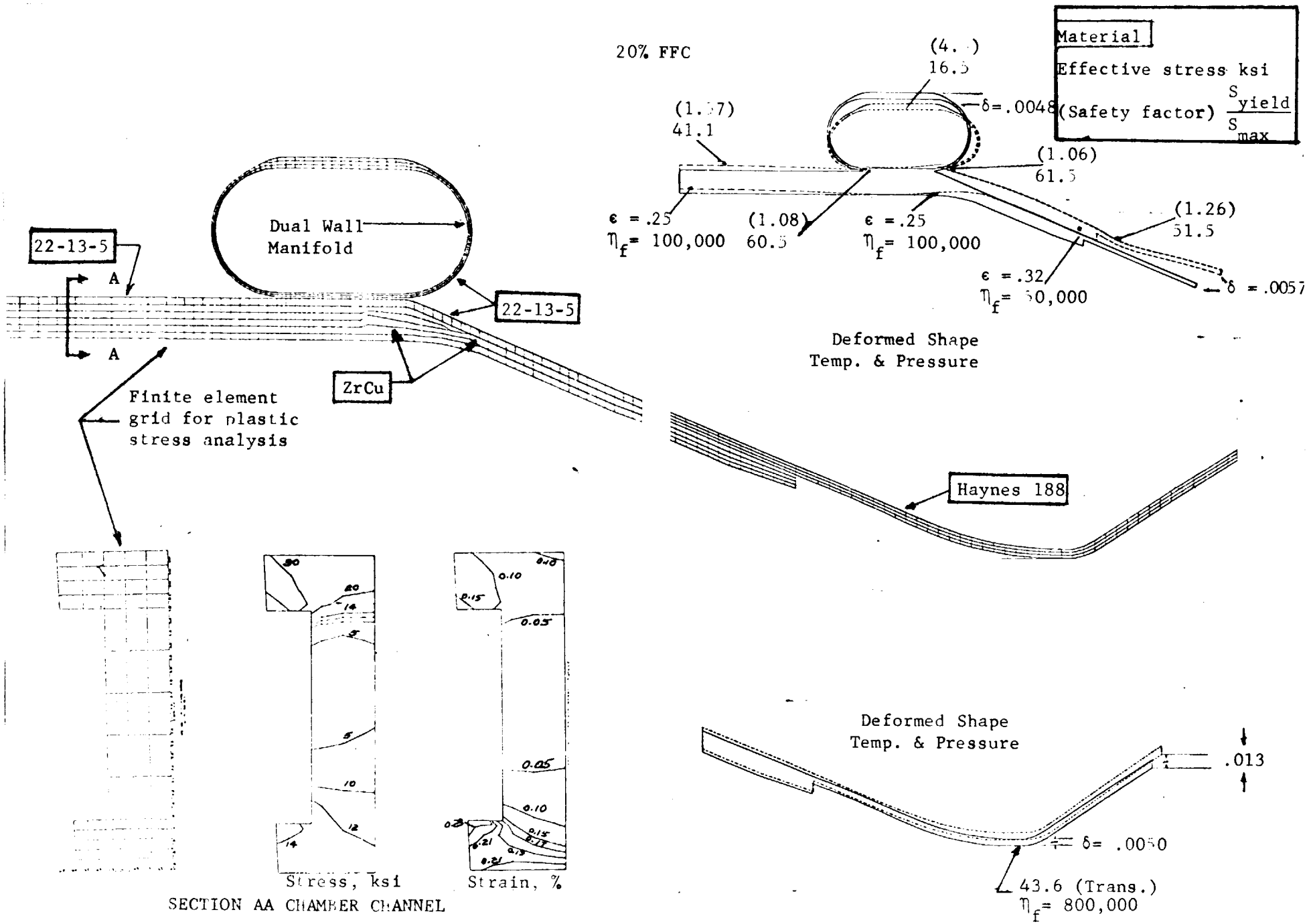


Figure 5

Film Cooled Chamber Structural Analysis
(Steady-State)

Structural (cont.)

Figure 6 provides a parametric analysis of the throat region fatigue life showing the influence of wall thickness and film cooling flow percentage.

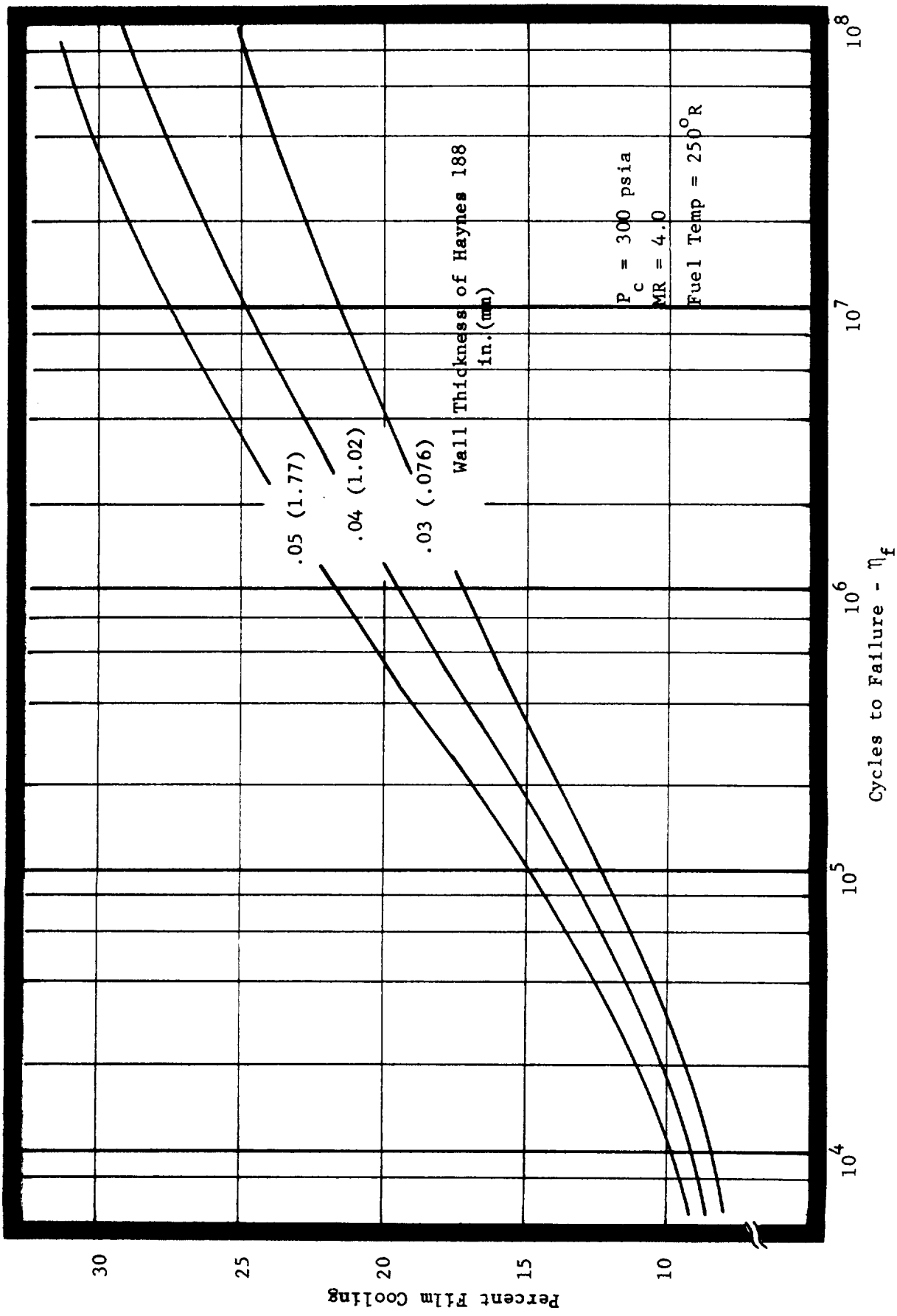

Percent Film Cooling
30
25
20
15
10
10^4
10^5
10^6
10^7
10^8
Cycles to Failure - η_f
Throat Region Fatigue Life
Wall Thickness of Haynes 188
in.(mm)
.05 (1.77)
.04 (1.02)
.03 (.076)
P_c = 300 psia
MR = 4.0
Fuel Temp = 250°R

Figure 6